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FEASIBILITY STUDY OF CRUISE FAN PROPULSION SYSTEMS AND ASSOCIATED POWER TRANSFER SYSTEMS FOR COMPOUND/COMPOSITE AIRCRAFT

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September 1967

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-336(T)
VERTOL DIVISION
THE BOEING COMPANY
MORTON, PENNSYLVANIA

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This report has been reviewed by the U. S. Army Aviation Materiel Laboratories and is considered to be technically sound. The report , which was prepared under Contract DA 44-177-AMC-336(T), presents an evaluation of cruise fans as potential propulsion systems for future compound/ composite aircraft.

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ABSTRACT

This report examines the feasibility and relative merit of five gas- and shaft-coupled cruise fan propulsion systems in various compound and composite aircraft configurations. A 1970 time period is assumed. Propulsion and airframe parameters are defined, and the results of an optimization process for maximum aircraft relative productivity in a fixed short-range transport mission are shown. Ranges of vehicle disc loading of from 5 to 11 pounds per square foot and of fan bypass ratio of from 3 to 12 are covered. The problems of rotor-fan power transfer systems are reviewed and possible solutions evaluated. Detailed weight data are given for optimum aircraft and propulsion system combinations. Convertible shaft-driven cruise fans in an integrated propulsion system are found to be particularly attractive; however, considerable work is required in development of power management systems.

FOREWORD

A feasibility study of cruise fan propulsion systems and associated power transfer systems for compound/composite aircraft was conducted by The Boeing Company, Vertol Division, for the U.S. Army Aviation Materiel Laboratories under Contract DA 44-177-AMC-336(T). The study was directed by the Project Manager, Mr. John J. Schneider, Manager of Vertol Division's Advanced Design Group. Supervision of all phases of the study was the responsibility of Mr. Philip C. Prager, Project Engineer. The principal assisting engineer was Mr. F. H. Dean. The study was initiated on 3 February 1966 and completed on 28 October 1966.

This study was performed under the technical cognizance of Mr. James Gomez and Mr. Henry Morrow of the U.S. Army Aviation Materiel Laboratories.

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CONTENTS

Page

ILLUSTRATIONS

Figure		Page

Figure		Page

Figure		Page

Figure		Page

Figure		Page

Figure		Page

Figure		Page

Figure		Page

Figure		Page
132	Propulsion System 3 for Tandem Rotor Aircraft	301
133	Propulsion System 1a for Single Rotor Aircraft	303
134	Propulsion System 1b for Single Rotor Aircraft	305
135	Propulsion System 2a for Single Rotor Aircraft	307
136	Propulsion System 2b for Single Rotor Aircraft	309
137	Propulsion System 3 for Single Rotor Aircraft	311
138	Power Turbine Horsepower With Partial Admission Throttling Valves	321
139	Partial Admission Turbine Efficiency and Power	322
140	Partial Admission Tip-Turbine Fan Thrust and Speed	323
141	Turbine Velocity Diagrams for Variable Turbine Geometry	324
142	Turbine Performance With Variable Stator Vanes	325
143	Generalized Thrust-Horsepower Relationship for Convertible Cruise Fan	326
144	Incidence Angles and Velocities for Fan With Variable Inlet Guide Vanes	327
145	Incidence Angles and Velocities for Fan With Variable Stagger Rotor Blades	328
146	Generalized Thrust-Horsepower Relationship for Fan With Variable Inlet Diaphragm (Inlet Baffles)	329

Figure		Page

Figure		Page

TABLES

SYMBOLS

Symbol		Unit
AHP	Auxiliary horsepower	hp
APU	Auxiliary power unit	
AR	Aspect ratio	
A_V	Damage vulnerable area	ft^2
B, BPR	Bypass ratio	
b_t	Total number of rotor blades per aircraft	
c	Rotor blade chord	ft
C_d	Drag coefficient	
C_ℓ, C_L	Lift coefficient	
$\bar{c}_\ell$	Average lift coefficient	
CP	Center of pressure	
C_T	Rotor thrust coefficient	
D	Main rotor diameter	ft
DL	Download force in percentage of thrust	
D/L	Disc loading	
D_{CB}	Diameter of combining gearbox	in.
D_{CL}	Diameter of decoupling clutch	in.
D_{EB}	Diameter of engine gearbox	in.
D_{PTB}	Diameter of power turbine gearbox	in.
D_{RGB}	Diameter of rotor gearbox	in.

Symbol		Unit
D_T	Tail rotor diameter	in.
e	Oswald span efficiency factor	
EW	Aircraft empty weight	lb
f_e	Equivalent flat plate area	ft^2
F_n	Net thrust	lb
g	Acceleration due to gravity	32.2 ft/sec^2
GW	Aircraft gross weight	lb
H_{RGB}	Height of rotor gearbox	in.
HP_I	Installed power value	hp
$(HP_I)_f$	Installed power value at fan hub	hp
k, K	A constant value	
K_d	Static droop weight penalty factor	
L	Distance between rotors	in.
L_{CL}	Length of decoupling clutch	
L/D	Lift to drag ratio	
$(L/D_E)_{max}$	Maximum equivalent lift to drag ratio	ft
L_{DV}	Length of diverter valve	ft
L.F.	Symmetrical flight maneuver limit load factor	
L_{GG}	Length of gas generator	ft
L_{PT}	Length of power turbine assembly	ft
L/PU	Lift/propulsion-unloaded aircraft type	
L_R	Distance between rotor shaft centers	ft

Symbol		Unit
ℓ_t	Tail arm	ft
L_X	Length of rear ramp well	ft
M	Mach number	
M_T	Rotor blade tip Mach number	
n	Number of rounds fired	
n	Ultimate flight maneuver load factor	
n_{CR}	Crash load factor	
Δp	Pressure drop	lb/ft^2 absolute
P_h	Hit probability	
P_k	Kill probability	
PNdb	Perceived noise level	decibels
PU	Propulsion-unloaded aircraft type	
P_X	Total transmission horsepower	hp
r	Distance along rotor blade axis	ft
R	Main rotor radius	ft
rpm	Revolutions per minute	rev/min
S	Total rotor disc area	ft^2
S, STD.	Standard ambient atmosphere conditions	
S_f	Fuselage wetted area	ft^2
SFC	Specific fuel consumption	lb/hp/hr
SHP	Shaft horsepower	hp

Symbol		Unit
SHP_{CB}	Shaft horsepower at combining gearbox	hp
SHP_{EB}	Shaft horsepower at engine gearbox	hp
SHP_{PT}	Shaft horsepower at power turbine	hp
SHP_{RGB}	Shaft horsepower at rotor gearbox	hp
SL	Sea level altitude	ft
S_{T_H}	Horizontal tail area	ft^2
S_{TV}	Vertical tail area	ft^2
S_W	Wing area	ft^2
T_3	Compressor exit temperature	°R
T_4	Gas generator turbine-inlet temperature	°R
T_f	Fan thrust per aircraft	lb
t/c	Thickness ratio; thickness/chord	
TOGW	Take-off gross weight	lb
TSFC	Thrust specific fuel consumption	lb/hr/lb
V	Aircraft forward speed	knots
V_D	Limit dive speed	knots
V_{max}	Aircraft maximum speed	knots
V_T	Rotor tip speed	ft/sec
W_a	Weight airflow	lb/sec
W_a'	Uncorrected weight airflow	lb/sec

Symbol		Unit
$W_a{}^*$	Sea level static maximum airflow	lb/sec
W_b	Rotor blade loading	lb/ft^2
W_{BS}	Body group basic structure weight	lb
W_{CF}	Weight of cruise fans or fan engines	lb
W_G	Gross weight	lb
W_{HT}	Horizontal tail weight	lb
W_{MR}	Main rotor weight	lb
W_R	Weight of one rotor	lb
W_T	Weight of tandem cascade thrust vectoring system	lb
W_{TG}	Tail group weight	lb
W_{TR}	Tail rotor weight	lb
W_w	Wing weight	lb
W_x	Weight of fuselage and contents	lb
X	Rotor disc overlap	ft
α_{PNF}	Rotor plane of no feathering angle	deg
Δ	Change from basic value	
Λ	Sweep angle	deg
η	Efficiency	
η_f	Fan efficiency	
η_p	Propulsive efficiency	
η_T	Transmission efficiency	
η_{CB}	Coupling or combining gearbox efficiency	

Symbol		Unit
η_{CR}	Crash load factor	
η_{EB}	Engine gearbox efficiency	
η_{PT}	Power turbine efficiency	
η_{PTB}	Power turbine gearbox efficiency	
η_{RGB}	Rotor gearbox efficiency	
η_{TRS}	Tail rotor drive system efficiency	
ρ	Air density	$slugs/ft^3$
ρ_o	Sea level standard air density	$slugs/ft^3$
σ	Air density ratio = ρ/ρ_o	
σ_R	Rotor solidity	
μ	Rotor advance ratio	

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SUMMARY

In February 1966 the United States Army Aviation Materiel Laboratories awarded The Boeing Company, Vertol Division, Contract DA 44-177-AMC-336(T) to conduct a research program to evaluate cruise fans as potential propulsion systems for future compound/composite aircraft. The study involved a 9-month effort. Consideration of both single and tandem rotor configurations of propulsion-unloaded compound aircraft, lift/propulsion-unloaded compound aircraft, and stowed rotor composite aircraft was included. Each of these six aircraft combinations was matched with the five propulsion systems listed below and shown in Figure 7 through 11.

1. System 1a, an integrated gas-driven tip-turbine cruise fan system.

2. System 1b, an integrated gas-driven hub-turbine cruise fan system.

3. System 2a, an integrated shaft-driven remote cruise fan system.

4. System 2b, an integrated shaft-driven convertible fan engine system.

5. System 3, an independent propulsion system.

In developing these 30 optimally matched systems, 3 disc loadings in a range of 5 to 11 psf, and 4 bypass ratios in a range of 3 to 12 were scanned. Because of the broad scope of the study, which involved 360 aircraft/propulsion design points, detailed designs of each of the optimum aircraft could not be developed; however, major characteristics sufficient for comparative evaluation purposes were identified.

Analysis of 1970 cruise fan engines led to the selection of a basic compressor pressure ratio of 13 and a sea level statistical standard day turbine-inlet temperature of 2460°R. Based on engine parametric studies and on these gas generator design parameters, propulsion system weight and performance data were developed.

Airframe scaling factors were also determined for the various aircraft types. The five propulsion systems were then integrated with each airframe configuration. The engines used for

vertical lift were sized, based on a 6000-foot, 95°F hover requirement. For each disc loading and bypass ratio, required gross and empty weights and aircraft cruise speeds were determined for the design 6000-pound payload, 100-nautical-mile radius sea level mission. Finally, the optimum characteristics of each of the 30 aircraft/propulsion system combinations were determined based on a relative productivity criteria. Disc loadings of optimum aircraft were generally on the high side of the study range. Optimum fan bypass ratios were generally higher in integrated propulsion systems and lower for independent system installations.

Analyses of power transfer systems indicated that variable turbine inlet arches are the most logical choice for power management in the gas-driven systems. For shaft-driven systems, variable inlet guide vanes were selected to effect fan power management; however, variable-pitch fan rotor blades had to be eliminated only for lack of engine manufacturer data.

Good feasibility of integrated propulsion systems for lift/propulsion-unloaded and composite aircraft was indicated, but the potential of these systems for propulsion-unloaded aircraft was questionable. Integrated systems appeared more desirable than independent systems. Propulsion systems providing the best relative productivity were the tip-driven gas coupled and the convertible fan shaft types. The convertible fan system was considered superior, however, in ease of installation, maintainability, and vulnerability.

Use of independent cruise fans for thrust assistance during hover flight increases mission relative productivity, but only if little mission hovering time is involved. In addition, severe forward area operational problems exist for such a system.

CONCLUSIONS

General

1. Integrated cruise fan propulsion systems are feasible for use with lift/propulsion-unloaded and composite aircraft.

2. The feasibility of integrated cruise fan propulsion systems for propulsion-unloaded aircraft is questionable. The study raises questions as to potential tradeoffs against advanced pure helicopters.

3. Integrated cruise fan propulsion systems, in general, appear to be more desirable than independent systems.

4. Integrated convertible cruise fans appear to be far superior to other systems in terms of major airframe design integration and environmental factors.

5. Preferred methods of power management for integrated propulsion systems involve fan variable inlet guide vanes for shaft-driven configurations and variable admission arches for gas-driven arrangements.

Propulsion-Unloaded Aircraft

1. No specific type of integrated propulsion system makes a clearly superior aircraft. Changes in fan configuration and basic parameters have little effect on overall aircraft characteristics. Fan thrust requirements are minor in practical cases.

2. Use of independent propulsion systems results in poorer aircraft characteristics than use of integrated systems, assuming equal aircraft speed.

3. Attempts to increase significantly the speeds of independent system aircraft above those of the integrated systems are not productive.

Lift/Propulsion-Unloaded Aircraft

1. At equal speeds, aircraft with integrated propulsion systems have characteristics superior to those of independent system aircraft.

2. Aircraft relative productivity appears marginally (2 to 3 percent) better than the other integrated propulsion systems where tip-driven gas-coupled fan systems are employed.

3. Independent system aircraft can be made competitive with integrated system aircraft in relative productivity only if their cruise speed is much higher; however, this results in a significant weight penalty.

Composite Aircraft

1. For equal aircraft speeds the convertible fan propulsion system is slightly (2 to 5 percent) superior in mission relative productivity to both the independent and other integrated propulsion systems.

Thrust-Vectored Fan Installations

1. Severe forward-area operational problems as well as difficult vehicle design integration problems are apparent with use of thrust-vectored fans in independent propulsion systems.

2. Thrust-vectored fans greatly improve relative productivity over equivalent nonvectored independent systems if a mission with little or no hovering time is considered. With increases in hover time, this superiority would reduce quickly.

RECOMMENDATIONS

Based on the results of the present study and the problem areas defined, the following recommendations are made concerning application of cruise fans to compound/composite aircraft:

1. Design, development, and testing of prototype integrated fan propulsion systems should be pressed forward in both convertible-type shaft systems and gas-coupled systems.

2. Power transfer components of these systems should be developed and further tested. Of particular interest is a full-range test (0 to maximum thrust) of variable inlet guide vanes and development of low-loss power turbine variable inlet arches for operation at high temperatures.

3. A feasibility and design study of a high-bypass-ratio cruise fan incorporating variable stagger blades should be conducted. Tradeoff analyses should be included comparing it to the systems considered in the present study.

4. A detailed analysis should be conducted making proper trade-offs between propulsion-unloaded compound aircraft with integrated propulsion systems and advanced design pure helicopters.

5. A detailed design study of two of the aircraft propulsion system combinations judged to be optimum in this study should be conducted to establish in detail the relative characteristics of the combinations. It is suggested that this study be conducted on either single rotor or tandem rotor lift/propulsion-unloaded aircraft incorporating a shaft-driven convertible cruise fan system and a tip-driven gas-coupled fan system.

6. Results of this study should be compared with the results of a similar study of the same aircraft with propeller-equipped propulsion systems.

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INTRODUCTION

It has long been recognized that the helicopter, with its relatively low disc loading, provides the maximum efficiency in hovering lift and low-speed transportation. The need for increased speed and range has caused many diverse programs to be instigated for the development of high-speed and increased lift/drag capability of rotary wing aircraft. Improvements in aerodynamic efficiency, propulsive efficiency, simplicity, and reliability will bring a realization of the full potential of this type of VTOL system.

The inherent advantages of low disc loadings (in the 10 psf range) are primarily apparent from the operator's viewpoint. The ability to autorotate safely in case of power failure and low downwash velocities, as well as low installed power, seems to generate a natural progression of rotary wing aircraft configurations to meet higher speed requirements. By advancing from the simple helicopter to the thrust-augmented compound helicopter (propulsion unloading only), increased speeds of at least 40 knots may be realized at the same vibration level. The compound helicopter with both lift and propulsion unloading gives even higher speeds and range. The composite aircraft with its airplane-like qualities in the cruise mode is especially attractive for certain missions.

In addition to obvious improvements (aerodynamic efficiency, etc.) in the rotor systems. the need for a propulsion system compatible with the compound/composite configurations requires investigation of the basic requirements for the propulsion system and therefore development of the optimum system to enable fulfillment of the potentials for advanced rotary wing aircraft.

In many previous investigations of compound configurations, shaft-driven propeller systems with their associated problems of aerodynamic flow influences, size (diameter) and location, complexity and large weight penalties left much to be desired. Initial studies have led to the conclusion that high bypass ratio cruise fans would be of interest for propulsion unloading of helicopters as well as forward thrust for composite aircraft.

Cruise fan powerplants may take a variety of forms such as concentric front fans, aft fans, remote shaft-driven shrouded fans, etc. Early in the development of aircraft gas turbines, it became apparent that a turbofan could provide the bridge in

propulsive efficiency between the propeller and the turbojet in the speed ranges of from 200 to 500 knots.

For many reasons, development of the turbofan engine was slow; however, in recent years the turbofan engine has received an increasing amount of interest in both commercial and military applications. Operational results in commercial airline service have been extremely favorable.

Over the past 10 years engine manufacturers have developed tip-turbine, high-bypass ratio, remote cruise fans and shaft-driven, shrouded cruise fans. A more recent development has been convertible engine configurations and associated power transfer methods to provide convertible engine characteristics required for compound/composite aircraft. Much research and testing have shown that all the cruise fan systems appear attractive for use in compound/composite configurations. However, there is need for an evaluation of the potential of these cruise fan powerplants both in an integrated propulsion system and in an independent thrust producer.

Recognizing this potential, the USAAVLABS funded two parametric design studies of cruise fan systems in 1964 to establish the technology base needed for the compound/composite cruise fan feasibility studies described in this report.

It is the purpose of this study to investigate compound/composite aircraft configurations to determine the propulsion system requirements and how best to meet these requirements with systems using high-bypass-ratio cruise fans. The potential of these cruise fan powerplants is evaluated both in an integrated propulsion system and in an independent thrust producer. Particular attention is paid to the power transfer requirements of each system, since the feasibility of these systems is directly related to the ability to manage the power split from hover through maximum cruise flight. Single and tandem rotor helicopters with varying disc loadings combined with both gas-driven and shaft-driven propulsion systems with various bypass ratios are analyzed, resulting in a matrix of optimum aircraft for comparison.

Since the primary objects of this study were the analysis and the comparison of cruise fan propulsion systems, no major attempt was made to arrive at optimum aircraft configuration design solutions for the various types. In certain cases, therefore, aircraft configurations may not include the most

desirable airframe features. It is not considered, for instance, that the composite aircraft shown in the study have the optimum means of blade fairing for high-speed flight. However, certain alternatives of design choice, such as retractable rotor transmissions, were considered less attractive. Since it was not practical to attempt invention of new design concepts during the course of the study, configurations were selected which appeared readily amenable to scaling and for use as a proper background to the main purpose of the study.

REVIEW AND ANAYLSIS OF CRUISE FAN PROPULSION SYSTEMS

Parametric investigations of high-bypass-ratio cruise fans conducted for the U.S. Army Aviation Materiel Laboratories were documented in References 1 and 18. Included in those studies were the variables of fan pressure ratio, bypass ratio, turbine-inlet temperature, and cycle pressure ratio for both mechanical and gas-coupled drive systems. This section of the report is devoted to the results of an analysis of this data, data from other engine manufacturers, data generated by the contractor, and the development of a meaningful summary of this data. This summary includes the selection of the basic cycle parameters, turbine-inlet temperature, and pressure ratio to be used during the integration part of the study. Following this selection, the available data was compiled and presented in a form applicable to the present study. The data presented is for a sea level static standard day turbine-inlet temperature of 2460°R and a compressor pressure ratio of 13; therefore, universal application of these results and trends is not recommended.

THERMODYNAMIC ANALYSIS

The gas generator design-point specific horsepower (gas generator available output horsepower divided by the gas generator airflow) is primarily a function of turbine-inlet temperature and increases with increasing temperature. Conversely, the specific fuel consumption (SFC) is primarily a function of the engine cycle pressure ratio and decreases with increasing pressure ratio. Therefore, there is no optimum engine pressure ratio which produces a minimum specific fuel consumption; but for each turbine-inlet temperature there is a maximum specific horsepower at each pressure ratio, although it is not sharply defined. The data obtained was formulated using representative assumed values of component efficiencies, pressure losses, and cooling airflows typical of 1970 state-of-the-art turbomachinery technology.

In the cruise fan study reported in Reference 1, the maximum thrust at sea level static conditions from the gas generator nozzle and fan combined was shown to occur when the fan turbine power was approximately 85 percent of the available power, after losses due to turbine efficiency and pressure drops were subtracted from the gas generator specific horsepower, leaving 15 percent of the kinetic energy in the fan turbine exhaust

flow. As a result, for a constant value of specific horsepower the fan pressure ratio for maximum thrust was a function of the fan bypass ratio. The cruise fan thrust characteristics verified that this choice of optimum design fan pressure ratio as a function of bypass ratio resulted in maximum cruise thrust.

Parametric Data Sources

In Reference 1, fan performance (thrust and thrust specific fuel consumption), weight, and dimensional data were presented for several cruise fan systems including direct-coupled concentric front fans, remote shaft-coupled fans, and remote gas-coupled tip-turbine fans. This study used a gas generator design pressure ratio of 13 for all the fan systems, and data was generated for families of parametric engines with design turbine-inlet temperatures of from 2160° to 2760°R and bypass ratios of from 2.6 to 11.6. At the design point of the front fan propulsion system, the fan and gas generators were matched to produce maximum gas generator rpm and turbine-inlet temperature. Because of the supercharging front fan and the higher inlet temperature to the compressor, referred rpm was less than that for the unsupercharged compressor, and the gas generator pressure ratio was less than 13 at the sea level static design point of the front fans.

In a similar study (Reference 18), performance, weight, and dimensional data for cruise fan systems, including direct-coupled concentric front fans, remote shaft-coupled fans, and remote gas-coupled axial-turbine fans, were developed. A gas generator design pressure ratio of 10 was selected, engines with gas generator pressure ratios of from 6 and 14 were added for comparison purposes, and bypass ratios of from 2 to 12.5 were used.

References 7 and 27 documented a library of performance and weight data for families of parametric cruise fan engines, including concentric front fans and remote gas-coupled tip-turbine fans. The design-point cycle parameters for these engines were engine pressure ratios of from 8 to 20, turbine-inlet temperatures of from 2200° to 2800°R, and bypass ratios of from 1 to 9. A statistical correlation was made of the weights of many production and study fan engines as a function of the basic cycle parameters.

In addition to these sources of engine data, Reference 20 provided performance and weight data for direct-coupled concentric front fans with engine design pressure ratios of from 15 to 30, turbine-inlet temperatures of from 2660° to 2960°R, and bypass ratios of from 2 to 16. Reference 16 provided performance and weight data for a gear-coupled concentric front-fan engine, which provided for declutching the fan and converting to shaft horsepower output or combining thrust and shaft power (convertible fan engine).

<u>Analysis of Fan Data</u>

Typical component efficiencies and pressure losses used in some of the various studies to generate fan performance data are summarized in Table I.

In analyzing the fan performance data from the various manufacturers, thrust specific fuel consumption was plotted as a function of specific thrust (thrust divided by design-point gas generator airflow) for different engine pressure ratios. The data from Reference 27 was plotted as a basis for comparison because it was the most extensive. Curves were plotted for static maximum thrust and maximum thrust at a flight Mach number of 0.3 at sea level. Against the background of this data, the performance of the engines in References 1 and 18 was plotted, and the trends of the Reference 27 data with turbine-inlet temperature, bypass ratio, and engine pressure ratio were used for interpolating and extrapolating thrust and thrust specific fuel consumption to obtain values of these parameters corresponding to even bypass ratios.

The thrust and thrust specific fuel consumption for concentric front fans, remote shaft-coupled fans, and remote gas-coupled fans with a gas generator design pressure ratio of 13 and a turbine-inlet temperature of 2460°R were cross-plotted as a function of bypass ratio. Typical variations in performance for concentric front fans and remote gas-coupled fans are summarized in Figures 1 and 2. For concentric front fans (Figure 1) slight differences in fan performance can be partly attributed to differences in fan efficiency assumptions. In Figure 2 the differences in thrust of the remote tip-turbine fans appeared to be the result of differences in duct loss assumptions. Generally, however, the performance data from References 1, 18 and 27 appeared to be consistent and in reasonable agreement. Furthermore, the component efficiency and pressure-drop assumptions for this data were well documented, making this performance data acceptable for use in the present study.

TABLE I

TYPICAL COMPONENT EFFICIENCIES AND PRESSURE LOSSES IN CRUISE FAN PROPULSION SYSTEMS

	Remote Gas-Coupled Fans		Concentric Front Fans		Remote Shaft-Coupled Fans
	Case		Case		Case
	1	2	1	2	1
Supercharging Fan η_f	-	-	0.90	0.85	-
Intercompressor Δ_P loss	-	-	3%	*	-
Compressor $\eta_{polytropic}$	*	0.90	*	0.90	*
Cooling Air Bleed Turbine	$f(T_4)$	2%	$f(T_4)$	2%	$f(T_4)$
Combustor Δ_P	$f(P.R., T_4-T_3)$	*	$f(P.R., T_4-T_3)$	*	$f(P.R., T_4-T_3)$
Combustor η	*	0.98	*	0.98	*
Gas Generator Turbine $\eta_{adiabatic}$	0.90	0.91	0.90	0.91	0.90
Interturbine Δ_P loss	-	-	1%	*	1%
Remote P.T. Duct Δ_P loss	5%+5%	5%	-	-	-
Power Turbine $\eta_{adiabatic}$	0.87	0.87	0.90	0.91	0.90
Power Turbine η_{gear}	-	-	-	0.995	0.97
Exhaust Nozzle Δ_P loss	1%	*	1%	*	4%
Inlet Ram Recovery	*	100%	*	100%	*
Fan $\eta_{polytropic}$	0.90	0.85	0.90	0.85	0.90
Fan Duct Δ_P loss	1%	0	1%	0	1%
Fan Nozzle Vel Coefficient	*	*	*	0.98	*

* Not Available

Selection of Cycle Parameters

As previously discussed, the gas generator specific horsepower is primarily a function of the turbine-inlet temperature. Increased turbine-inlet temperature results in increased specific horsepower, therefore requiring a smaller airflow and a smaller engine for the same power. However, to operate at this increased turbine-inlet temperature may require cooled turbines, which means an increase in engine weight and complexity; it also results in a poorer thrust specific fuel consumption for the cruise fan and therefore in a greater fuel weight. An increase in engine pressure ratio results in an improved specific fuel consumption and in a smaller fuel weight for the same power or thrust, but this increase in pressure ratio must be accomplished by increasing the size of the compressor and possibly the turbine.

The selection of the gas generator pressure ratio and turbine-inlet temperature to be used in the present study required tradeoff studies of the fuel plus propulsion system weights involved. The cruise fan studies in Reference 1 included a mission analysis for a compound helicopter mission much like the one outlined in the present s[illegible], with the results presented in the form of fuel weight plus propulsion system weight (see Figure 3).

Figure 4 shows the results of a similar calculation with front-fan and remote tip-turbine fan data from References 7 and 27. Figure 3 indicated an optimum turbine-inlet temperature for each bypass ratio with the various cruise fan systems to be generally in the range of from 2400° to 2600°R. Figure 4, however, showed fuel plus propulsion system weight generally increasing with increasing turbine-inlet temperature.

To assist in the selection of a turbine-inlet temperature for the engines of the present study, the state of the art of turbine technology with regard to turbine-inlet temperature was plotted in Figure 5. This data, obtained from various engine manufacturers, together with the information presented in Figures 3 and 4, suggests 2000°F (2460°R) as a reasonable estimate for the design turbine-inlet temperature for 1970 gas generators. Furthermore, it appears that such turbines will require a minimum of turbine cooling, considering the advances in materials technology.

A gas generator pressure ratio of 13 was selected for the Reference 1 cruise fan study, while a more conservative pressure ratio of 10 was selected for the Reference 18 study. The higher the pressure ratio, the better the specific fuel consumption. Reference 1 demonstrated that the fuel plus propulsion system weight for the compound helicopter mission with remote tip-turbine fans decreased with an increase in gas generator pressure ratio to 20. However, achieving higher pressure ratios for engine sizes envisioned in the present study might result in extremely small rear stages in the compressor, with a resultant performance penalty. Also, higher gas generator pressure ratios would require more compressor stages, which would create shaft problems with the front fan designs. Higher gas generator pressure ratios would diminish the performance advantages of the supercharging front fans, while very low gas generator pressure ratios would penalize the remote fan performance. The above reasoning led to the selection of a gas generator pressure ratio of 13 for the present study.

Engine Lapse-Rate Data

Since the propulsion system must provide enough power to the rotors to permit hover at 6000 feet on a 95°F day, the relationship between that power and sea level static maximum power had to be determined. In Figure 6 data points were plotted from various engine manufacturers' production and development engine specifications in order to establish that relationship for the present study.

Curves were plotted in Figure 6 for engines which are flat-rated as well as for engines which are not. The flat-rated engine would produce substantially more power at high ambient temperatures, at sea level and altitude. The engine manufacturer would design the fuel control for such an engine in order to permit operation up to the limiting fuel flow, with a turbine temperature limitation somewhat greater than the sea level standard design turbine-inlet temperature.

Many of the more advanced shaft engines have incorporated this improved hot-day performance to some degree. The engine manufacturers have configured these engines to produce increased power at high ambient temperatures in order to make their hot-day hover performance much more attractive. Based on these correlations of manufacturers' data, some improvement in hot-day performance of the engine was assumed, and maximum

shaft horsepower for 6000 feet, 95°F, was taken to be 0.78 of that for sea level standard.

Conceptual Designs

Both integrated and independent fan-propulsion systems were evaluated. The integrated systems were (1a) remote gas-coupled tip-turbine, (1b) remote gas-coupled axial-turbine, (2a) remote shaft-driven, and (2b) convertible concentric shaft-driven. The independent system was a concentric front fan.

Remote Gas-Coupled Tip-Turbine Fan Systems

The source of power for the remote gas-coupled tip-turbine fan propulsion system (see Figure 7) was two gas generators with a design pressure ratio of 13 and a turbine-inlet temperature of 2460°R. At the exhaust flange of each gas generator, an adapter section of duct was connected to a diverter valve sized for a flow Mach number of 0.3. Flappers were located in each discharge leg of the diverter valve to shut off the flow to either the fan or the remote power turbine. From each diverter valve, the hot gas flow was ducted to a common, remote power turbine, as well as to the respective fan.

Variable admission arches, in which each segment of the power turbine inlet annulus was separately ducted with a gate valve for flow modulation, controlled the amount of flow into the power turbine. An exhaust diffuser minimized the turbine downstream pressure and residual kinetic energy, and maximized shaft horsepower.

In cruise operation, the hot gas was diverted either in part (compound aircraft) or wholly (composite aircraft) to the tip turbine, where variable admission arches were also used to control the amount of flow. The tip-turbine rotor blades were extensions of the cruise fan blades with a shroud band dividing the primary and secondary flows.

Remote Gas-Coupled Axial-Turbine Fan Systems (Reference 18)

The remote gas-coupled axial-turbine fan system (see Figure 8) was the same as the tip-turbine fan system except for the fan configuration: a multistage turbine was located within the center body nacelle of the cruise fan and coupled directly to the fan. As with the tip-turbine system, fan-turbine flow control was accomplished with variable admission arches.

Remote Shaft-Coupled Fan Systems

The source of power for the remote shaft-coupled fan system (see Figure 9) was standard turboshaft engines with a curved exhaust diffuser, a rear-drive shaft, and a rear-mounted gearbox. This concept differs somewhat from that discussed in Reference 1, where the power turbine was followed by an exhaust nozzle, so that the combined fan and gas generator-power turbine thrust would be nearly optimum. However, for the installations proposed for these compound/composite aircraft, it was not practical to consider the use of the gas generator power turbine exhaust gases for thrust. Therefore, an open exhaust diffuser was substituted to minimize the exhaust kinetic energy and thrust to maximize the fan power and thrust.

The gas generator power turbine gearbox was shaft-coupled to the rotor system coupling gearbox. In addition, the remote fan was shaft-coupled to the gas generator power turbine gearbox with a decoupling clutch to cut off the power to the cruise fan in the hover mode. In the cruise mode the clutch was engaged, and the amount of fan power and thrust was regulated by variable fan-inlet guide vanes which controlled the amount of fan airflow.

Convertible Concentric Front Fan Systems

The convertible fan concept (see Figure 10) consisted of a gas generator with a free power turbine gear-coupled to a front fan, a power shaft from the side of the engine driven off the same gear train, and a decoupling clutch between the power takeoff shaft and the fan. With the clutch engaged in the cruise mode, variable fan-inlet guide vanes regulated the fan airflow, fan power, and thrust, and the remaining free turbine power was available for the output power shaft. In the hover mode the fan clutch was disengaged, providing maximum power to the output shaft.

Independent Concentric Front Fan Systems

The final propulsion system consisted of independent concentric cruise fan (see Figure 11) and independent turboshaft engines to provide the required rotor power. The turboshaft engines had a curved exhaust diffuser with a rear drive shaft and a rear-mounted gearbox which was shaft-coupled to the rotor system coupling gearbox. The cruise fans were direct-coupled concentric front fans.

PROPULSION SYSTEM PHYSICAL DATA

Dimensional Data

Remote Gas-Coupled Tip-Turbine Fan Systems. The maximum length of the gas generator for these propulsion systems was determined by a correlation of manufacturers' production and brochure engine data, while the maximum diameter was taken from Reference 1 data for the fan turbine for remote shaft-coupled fans, which reflected the assumption of higher tip speeds and smaller diameters. This data is presented in Figure 12.

The duct and diverter valve diameters were based on a flow Mach number of 0.3. The dimensional relationships of the diverter valves were taken from Reference 11. These dimensions are presented in Figure 13.

The diameters of the remote power turbine were calculated for a turbine exit Mach number of 0.45 and a diffuser exit Mach number of 0.2. These diameters proved to be consistent with manufacturers' advanced engines. Length dimensions were based on correlations for the inlet section from Reference 1 and typical curved exhaust diffuser dimensions. These dimensions are given in Figure 14. Tip-turbine fan dimensions are shown in Figure 15 (Reference 1).

Remote Gas-Coupled Axial-Turbine Fan Systems. The dimensional data for the gas generators, ducts, diverter valves, and remote power turbine was the same as that for the tip-turbine fans. The axial-turbine fan dimensional data (see Figure 16) was compounded from the data for geared concentric front fans in Reference 1 and other advanced study data.

Remote Shaft-Coupled Fan Systems. The maximum diameter and exhaust diameter of the turboshaft engine for the remote shaft-coupled fan system (see Figure 17) were calculated based on Mach numbers of 0.45 at the turbine exit and 0.2 at the exhaust flange and were checked with Reference 1 for consistency. Length dimensions resulted from a correlation of manufacturers' turboshaft engines modified with a curved exhaust diffuser. Fan dimensions were taken from a correlation of the data in Reference 1.

Convertible Concentric Front-Fan Systems. Dimensional data for the convertible front fans in Figure 18 was taken from a correlation of geared concentric front-fan data in Reference 1.

Independent Concentric Front-Fan Systems. Dimensional data for the turboshaft engines in the independent concentric front-fan system was the same as that in Figure 17. The data for the fans in Figure 19 resulted from a correlation of direct-coupled concentric front-fan data obtained from Reference 1.

Propulsion System Weights

Decreases in turbomachinery weight for 1970 state-of-the-art technology will come from several sources. Improved component design techniques will permit higher aerodynamic loading, less compressor and turbine stages, and smaller diameters. Advances in materials technology will allow higher stresses, higher wheel speeds in compressors and turbines, and consequently smaller diameters and lighter weight. Use of new materials, such as titanium in the compressor, will further reduce weight. In addition, careful attention to design details and fabrication techniques will effect even greater improvements.

Fan weight correlations demonstrating these anticipated weight improvements showed a 40 percent reduction in weights of the remote gas-coupled axial-turbine systems, a 15 percent reduction in weight of the remote shaft-coupled systems and a 15 percent reduction in weight of the direct-coupled concentric front fans. Front-fan weight resulting from this technology and material change is shown in Table II.

Although the weight data presented in Reference 1 was significantly lighter than for current technology, it was consistent with advanced design and development engine technology already exhibited by the manufacturer. Consequently, much of the weight data from Reference 1 has been incorporated in the present study.

TABLE II

FRONT-FAN WEIGHT DATA

Bypass Ratio	Specific Weight (lb/lb/sec)	
	Current Technology	Adv. Technology
3	4.60	6.15
6	3.70	5.02
9	3.25	4.55
Specific weight of front fans = weight/total airflow		

Powerplant Weights

Remote Gas-Coupled Tip-Turbine Fan Systems - The weight of the gas generator and associated controls and accessories for the tip-turbine fan propulsion system, plotted in Figure 20, was derived from the data in Reference 1, and checked for consistency with advanced engine technology. Diverter valve and duct weights (see Figure 21) were obtained from Reference 6, which correlated data for existing installations. Weights for the remote power turbine (see Figure 22) were produced by combining the power turbine weights for single-stage turbines from the geared front fans and remote shaft-coupled fans of Reference 1 with nacelle weights to simulate the power turbine inlet. Also, comparison with multistage remote power turbine weight data from advanced studies and interpolation for the required two-stage remote power turbine was made. The weight of the tip-turbine fan was derived from a correlation of Reference 1 data and is presented in Figure 23.

The fan limits indicated in Figure 23 at low gas generator airflow and bypass ratio are a result of mechanical constraints. The fan bearings are sized according to power requirements and establish an rpm limit. At low gas generator airflows, decreasing bypass ratio should be accompanied by decreased tip diameters to maintain a reasonable hub-tip radius ratio and increased rpm to maintain design tip speeds. Once the rpm limit is reached, the fan-tip diameter must remain constant to produce design tip speed and the hub diameter must increase as bypass ratio decreases. The result is an increase in the fan disc diameter at low bypass ratios. Figure 23 shows that for bypass ratios less than 6, the slope of the fan weight trend changes significantly (for example, the trend at bypass ratio of 4.18); and at the bypass ratio of 3, fan weight increases with decreasing airflow. At a bypass ratio of 3, tip-turbine fans with a gas flow less than 30 pounds per second were eliminated because the thrust/weight ratio was unreasonably low.

Remote Gas-Coupled Axial-Turbine Fan Systems - To generate remote axial-turbine fan system weight data, Reference 1 weights for power turbine, fan, and shaft for the direct-coupled front fan, modified slightly for the turbine inlet configuration, were used. This weight data proved to be consistent with other remote axial-turbine fan weight data for advanced study engines (see Figure 24). Weight correlations for the gas generator, ducts and diverter valve, and remote power turbine were the same as those for the remote tip-turbine

fan propulsion system.

Remote Shaft-Coupled Fan Systems - The weight data for the remote shaft-coupled fan propulsion systems (see Figure 25) was determined from a correlation of the weights of this system given in Reference 1. Small differences in gas generator-fan turbine gearbox weight can be attributed to differences in the gearbox, where larger gear reductions are necessary for the high-bypass-ratio fans.

Convertible Concentric Front-Fan Systems - A correlation of the weights of geared front fans from Reference 1 was the primary source of weight data for the convertible fan propulsion system. To this was added the weight of the clutch, which resulted from a correlation of manufacturer's data that assumed the clutch weight to be proportional to torque. An additional correction was added for variable fan-inlet guide vane weights. The total for the convertible fan is shown in Figure 26.

Independent Concentric Front-Fan Systems - To determine the weight of the turboshaft engines for the independent cruise fan propulsion system, the sum of the weights of the gas generator, fan turbine, shafts and bearings, and controls and accessories for the geared front fans, remote shaft-coupled fans (single-stage fan turbine), and the direct-coupled front fans (multistage fan turbine) was plotted from Reference 1. Data for manufacturers' advanced turboshaft engines was added, and a compromise curve was determined for a two-stage turboshaft engine (see Figure 27). Weights for the independent concentric front fans, plotted in Figure 28, were taken directly from a correlation of the data for direct-coupled concentric front fans in Reference 1.

Accessory Weights

Controls and engine accessories included in the gas generator and fan weights are as follows:

Gas Generator

1. Fuel supply including pump, filter, valves, plumbing, and hardware

2. Main fuel control, engine governor, CIT sensor, and stator actuators

3. Lube system including supply and scavenge pump, fuel-oil cooler, plumbing, tank, and hardware

4. Ignition system

5. Accessory drive gear case

6. Miscellaneous air piping, anti-icing, T_5 harness, etc.

Fan Systems

1. Power-turbine lube system

2. Gear-set lube system including air-oil cooler and tank

Gearbox, Transmission, and Shaft Weights

These systems are included in a later section of this report concerning system weight methodology and weight trends (page 120).

PROPULSION SYSTEM PERFORMANCE

Fan Performance

The fan performance data from various sources, plotted in Figures 1 and 2, illustrated a reasonable consistency between the two data sources for front fans and remote tip-turbine fans. Figure 29 presents bare engine specific thrust and thrust specific fuel consumption data for the different fan propulsion systems for sea level standard day inlet conditions, maximum power setting, and static and Mach number 0.3. The concentric front fan thrust specific fuel consumption is consistently better than that of the other systems because of the supercharging effect of the fan, an advantage that gradually disappears at higher bypass ratios as the fan pressure ratio becomes smaller. Another observation is that the performance of the revised remote shaft-coupled fan is significantly poorer than the others. This was a result of a configuration change necessitated by installation problems to minimize gas generator power turbine thrust and to maximize fan power and thrust, giving off-optimum fan thrust.

A fan drag analysis was performed to determine installation drag. The factors considered in installation drag were the

fan external shroud drag, the inlet drag, and the blowing drag, which is defined as the drag force experienced by the engine cowl when the total pressure of the fan nozzle is increased by the ram effect of the air at flight velocity.

Installed specific thrust and thrust specific fuel consumption have been plotted for each of the cruise fan propulsion systems for bypass ratios 3, 6, 9, and 12. The data was presented in the form of thrust specific fuel consumption versus normalized thrust, defined as installed thrust divided by sea level static maximum fan-turbine airflow, with parameters of Mach number. Figure 30 presents the installed specific thrust and the thrust specific fuel consumption for the remote gas-coupled tip-turbine fans. The fan performance was based on a 5-percent pressure drop in the tip-turbine manifold and scrolls and a 5-percent duct loss. The loss in the diverter valve and duct closely approximated this value. A simple correction factor (see Figure 31) corrected tip-turbine fan performance to produce remote axial-turbine fan performance, which accounted for an increase of 3 percent in fan-turbine efficiency.

Figure 32 presents the fan performance for the remote shaft-coupled fans, and the performance of the independent concentric front fans is shown in Figure 33. The difference in fan performance between the convertible fan and the concentric front fan was simply the losses in the gears. A correction factor for this has been plotted in Figure 34.

Shaft Power Performance

The sea level static maximum horsepower was calculated for the free power turbine of each of the cruise fan propulsion systems using the gas generator exit temperature, pressure, and power turbine efficiency obtained from Reference 1. For the remote power turbine of the gas-coupled remote fan systems, additional assumptions were made. Diverter valve pressure drop was assumed to be 4 percent diverted and 3 percent straight through, based on data in Reference 5. Also, duct and bend pressure drops were determined for a flow Mach number of 0.3 from data in Reference 10.

For the remote power turbine, two turbine stages were required. The turbine shaft power is shown in Figure 35 and the turbine rpm is presented in Figure 14.

For the fan turbine of the remote shaft-coupled fan system,

7-percent exhaust diffuser pressure loss and 98½-percent gear mechanical efficiency were assumed. This resulted in a specific horsepower at the rotor cross shaft of 171.1 horsepower per pound per second of fan turbine airflow and a specific fuel consumption of 0.411 pound per hour per horsepower at sea level static maximum power setting. Power turbine rpm and airflow were correlated by the relation (Reference 1)

$$\text{rpm} \times \sqrt{\text{Airflow (lb/sec)}} = 124{,}500$$

In calculating specific horsepower of the free turbine of the convertible fan engine, pressure losses and gear losses from Reference 1 were assumed. The power and specific fuel consumption are presented in Figure 36.

Calculations were made for the turboshaft engine of the independent front-fan propulsion system with an assumed 7-percent exhaust diffuser pressure loss. This engine required a two-stage power turbine. The turbine output specific horsepower for this system was 173.6 horsepower per pound per second of airflow, and the specific fuel consumption was 0.411 pound per hour per horsepower at sea level static maximum power setting. Power turbine rpm and airflow were correlated by the relation

$$\text{rpm} \times \sqrt{\text{Airflow (lb/sec)}} = 120{,}000$$

For all these shaft power turbines, the trends of power, specific fuel consumption, and airflow with Mach number and power setting were approximated by data from manufacturers' advanced turboshaft engines. These trends are shown in Figure 37.

COMPONENT TECHNOLOGY

Nozzle Configurations

In Reference 1, a two-position fan jet nozzle was used for the low-fan-pressure ratio, high-bypass-ratio fans to provide good fan efficiency over the operating range of flight Mach numbers. From some additional manufacturer's data on advanced shaft engines, the data shown in Figure 38 was plotted. This data illustrates that an exhaust nozzle sized for Mach number 0.3 would provide efficient operation over a wide range of Mach numbers. For the mission envisioned in the present study, efficient fan performance is required only over a narrow range of Mach numbers; therefore, it would not be practical to add

the complexity and weight of variable jet nozzles to the cruise fan propulsion systems.

Vertical Thrust Augmentation

Consideration has been given to the feasibility of vertical thrust augmentation in hover with the independent concentric cruise fan propulsion systems. Among the design concepts analyzed were tilting engines, thrust vectoring systems such as swivel nozzles, tandem cascades, and rotatable vanes, and a system of blocker doors with louvered nozzles for vertical thrust. A brief analysis was made of performance, weight, and complexity of each of these concepts.

The design and development of the swivel nozzle system was discussed in Reference 26. The velocity coefficient was stated to be 0.985 in the horizontal cruise position and 0.95 in the vertical vectored lift position. The tandem cascade-thrust vectoring system was documented in Reference 25. This system had a velocity coefficient of 0.926 through 90 degrees of turning, using flexible cambered airfoils (also considered were fixed camber vanes, flapped airfoils, and articulated airfoils). A concept of rotatable vanes proposed in Reference 21 used a transition section downstream of the fan and a fan turbine consisting of a 45-degree turning elbow duct with a rotatable cascade of 45-degree deflection nozzle vanes. In a full 180-degree rotation of this cascade, the flow direction can be altered from vertical to horizontal. The velocity coefficient of this system was said to be 0.985. No weights were available for this concept.

The system suggested in Reference 22 consisted of blocker doors which redirect the flow in the vertical thrust mode through variable louvered cascades in the underside of the engine nacelle. For the cruise mode, these blocker doors were retracted and the cascade louvers were closed to direct the flow in the normal horizontal direction. Performance calculations for such a configuration indicated extremely large losses due to turning the flow abruptly, even at reasonably small Mach numbers. When the flow is deflected, it is anticipated that the velocity coefficient for such a system would be less than 0.90.

All the thrust vectoring systems discussed above envisioned a long duct-fan configuration, rather than the short-duct arrangement of Reference 1 for the independent concentric cruise fans.

According to Reference 22, this constituted a 10-percent increase in bare engine weight and a 6- to 10-percent increase in installed weight. Although weight data for the thrust vectoring systems was somewhat limited, weight data was available for some systems. The weight of the tandem cascade configuration, including the rotating joint, was correlated by the relationship

$$W_T = 3.16 \times \sqrt{\text{total fan + engine airflow (lb/sec)}}$$

This relationship was shown in Figure 39, with additional data from Reference 4 for a swivel nozzle system and the system with blocker doors and louvered cascades. Although the systems were not identical to those previously discussed, the weights were considered to be representative.

When weight is considered, it is believed that the tandem-cascade system was the most promising thrust vectoring system, although performance losses were somewhat greater than with a swivel nozzle. As mentioned above, weights for the rotatable-vane system were unavailable, but they should also be quite low. In addition to vectoring systems, fan tilt can also be used for vertical thrust augmentation. Design integration and feasibility as well as system applicability must be considered in the use of these systems and are discussed later.

Results

The review and analysis of cruise fan propulsion systems as presented in parametric studies (References 1, 7, 18, 20, 27) provided the following results to be used in this cruise fan feasibility study:

1. Selection of 2460°R for design turbine-inlet temperature.

2. Selection of a compressor pressure ratio of 13.

3. Duct and diverter valve weights and sizes based on Mach number 0.3 and, therefore, related to the design turbine-inlet temperature of 2460°R.

4. Selection of 0.78 as the engine power lapse rate from sea level standard to 6000-foot altitude at 95°F.

5. Weight and dimensions for all propulsion systems based on a design turbine-inlet temperature of 2460°R and a

pressure ratio of 13 which reflect 1970 state of the art.

6. Fixed area nozzles.

Noise Evaluation

The noise levels of aircraft are increasingly important in applications and design of military and commercial aircraft. The major noise source of existing aircraft (except for rotor bang) and the anticipated majority of future aircraft is the propulsion system. Therefore, inherent noise characteristics of the five propulsion systems were considered as part of the development of component technology for the cruise fan propulsion system study.

Since the turbofan is a hybrid derived from turboprop and turbojet designs, it was decided that the proven concept of PNdb and perceived noise level be applied in evaluating the noise level of a wide variety of cruise fans and associated major noise producers under different operating conditions (Reference 13). This concept accounts for both the magnitude and the frequency content of a noise by a single number and thus permits relatively easy evaluation of otherwise voluminous and complicated information.

Specifically, the noise of gas generators, power turbines, bypass cruise fans and turboshaft engines was examined. Due to practical work limitations, the evaluation excluded rotor, gear, and transmission noise, which, except for rotor noise, should contribute relatively little to external noise characteristics. Another simplification has been made by first predicting the inherent propulsion system noise level only, without considering installation details of all 30 configurations. Installation details have been evaluated, for example, only for the tandem lift/propulsion-unloaded configuration. Possible future noise reduction due to, for example, rotor-stator spacing, inlet choking, or exhaust noise suppressing treatments could be accounted for when such techniques have advanced to the state of becoming practical for aircraft production.

Noise Prediction Methods - The sources of noise considered are compressor, power turbine, fan inlets and gas generators, primary and bypass cruise fans, and power turbine exhausts. The inlet-noise prediction is based on an experimentally derived relationship between mechanical tip speed and diameter of

compressors (fans) and the resulting noise level at the fundamental blade passage frequency (Reference 17). The exhaust noise predictions are based on jet parameters as verified by in-house jet engine exhaust noise data, and other data such as that of Reference 14. Although based generally on larger powerplants than those with which the present study was concerned, the method is considered to be accurate within ±5 db and has been verified on a small engine.

Some of the simplifications and assumptions are as follows:

1. Noise-level spectra were predicted over a frequency range of from 37.5 to 19,400 cycles per second.

2. Aircraft attitude and powerplant locations result in directivity indexes which affect the frequency distribution of the noise for a particular observer. However, this fact does not change the perceived noise levels shown.

3. Engine exhaust noise spectra are assumed to be similar in shape. The same was assumed for the inlet-noise distributions.

Results and Discussion - The inherent external noise characteristics of several types of propulsion systems are summarized in Figure 41. Data for the combined inlet noises of a propulsion system was separated from the exhaust noises of the various components. Also shown is the total noise of all sources, including inlet and exhaust noise.

Since different power requirements exist for various components of any one propulsion system depending on the flight profile, noise levels are shown for hover and cruise at bypass ratios of 3 through 12.

Figure 40 is an example of how the optimum combination of aircraft and propulsion system, from a total noise viewpoint, was obtained. The aircraft considered is the tandem lift/propulsion-unloaded configuration with each one of the propulsion systems at its optimum performance bypass ratio. After making allowances for installation into the airframe, comparison of the cross-hatched areas provided a ranked order of propulsion systems in regard to noise (refer to Table III).

TABLE III. COMPARISON OF NOISE LEVELS IN PROPULSION SYSTEMS

Propulsion System	B.P. Ratio	Rank (hover mode)
Remote tip-turbine fans (1a)	12	3
Remote axial-turbine fans (1b)	12	3
Remote shaft-driven fans (2a)	12	1
Convertible concentric front fans (2b)	9	2
Independent concentric front fans (3)	3	2

It should be pointed out that a difference of less than 3 PNdb between any two levels is barely noticeable. This makes propulsion system 2a the best of all applied to the tandem lift/propulsion-unloaded configuration; 2b and 3 second; and 1a and 1b worst. Since all systems are within 1 PNdb in cruise, they are considered to be equal in that mode.

A similar analysis of all 30 optimum aircraft propulsion systems was conducted; each was ranked according to the level of inherent external noise (refer to Table IV) considering annoyance (caused by high-frequency inlet noise) and detection (controlled by the amount of low-frequency exhaust noise). The ranking ranges from 1, which indicates the least annoying level (or least detectable), to 4, which indicates the most annoying level (or most easily detectable).

TABLE IV
RANKED ORDER OF PROPULSION SYSTEMS ACCORDING TO EXTERNAL NOISE LEVELS

Propulsion System	Annoyance		Detection	
	Hover	Cruise	Hover	Cruise
	Tandem Composite			
1a	2	-*	1	1
1b	2	-	1	2
2a	1	-	1	1
2b	1	-	2	2
3	1	-	1	3

TABLE IV - Continued

Propulsion System	Annoyance Hover	Annoyance Cruise	Detection Hover	Detection Cruise
		Single Composite		
1a	3	-*	1	1
1b	3	-	1	3
2a	2	-	1	1
2b	1	-	2	2
3	2	-	1	4
		Tandem LPU		
1a	3	-*	1	1
1b	3	-	1	2
2a	2	-	1	1
2b	1	-	2	2
3	2	-	1	3
		Single LPU		
1a	3	-*	1	1
1b	3	-	1	2
2a	2	-	1	1
2b	1	-	2	2
3	2	-	1	3
		Tandem PU		
1a	3	-*	1	1
1b	3	-	1	4
2a	2	-	1	2
2b	1	-	2	3
3	2	-	1	4
		Single PU		
1a	3	-*	1	1
1b	3	-	1	3
2a	2	-	1	1
2b	1	-	2	2
3	3	-	1	4

*All systems are essentially equal in cruise for this aircraft type.

The most significant trend observed with increasing bypass ratio is the reduction of cruise fan exhaust noise. The implication is that if inlet noise could be lowered sufficiently to the level where it is comparable to the exhaust noise, the choice of an optimum bypass ratio would always be 12. But inlet noise-level reductions in excess of 15 or 20 db are unlikely in the foreseeable future; thus, it appears that inlet noise would establish the total noise levels. However, this relationship changes with increased distance from the aircraft, for high-frequency inlet noise will eventually be absorbed by the atmosphere much more rapidly than low-frequency exhaust noise; therefore, considering detection, a propulsion system with a high bypass ratio is still the best choice.

In general, inlet-compressor and fan noise predominates at all bypass ratios for all propulsion systems with the exception of the convertible cruise fan system. During hover, the latter operates at relatively higher engine pressure ratios and, therefore, results in exhaust noise levels comparable to that system's inlet noise. With the exception of the convertible cruise fan system, the total noise level is always set by the high inlet noise of either cruise fans or power turbines (1a and 1b). Even after considering the effects of installation of the systems in aircraft, the rank order relationship within the tandem lift/propulsion-unloaded group is unchanged.

To arrive at an estimate of the effect of enclosing the power turbine in an aircraft structure, it was assumed that noise would be reduced due to sound transmission loss through an 0.032-inch aluminum structure according to the mass law. Stiffness, resonance, and wave-coincidence effects were neglected in this simplified assumption. However, much of the noise reduction that could be obtained by a complete enclosure is negated, since the sound radiates down through the open exhaust ducts.

It should be noted that rotor noise levels, which were not accounted for in these cruise fan noise studies, could be significantly higher than either propulsion system inlet or exhaust noise and, therefore, should be included in a comprehensive operational analysis.

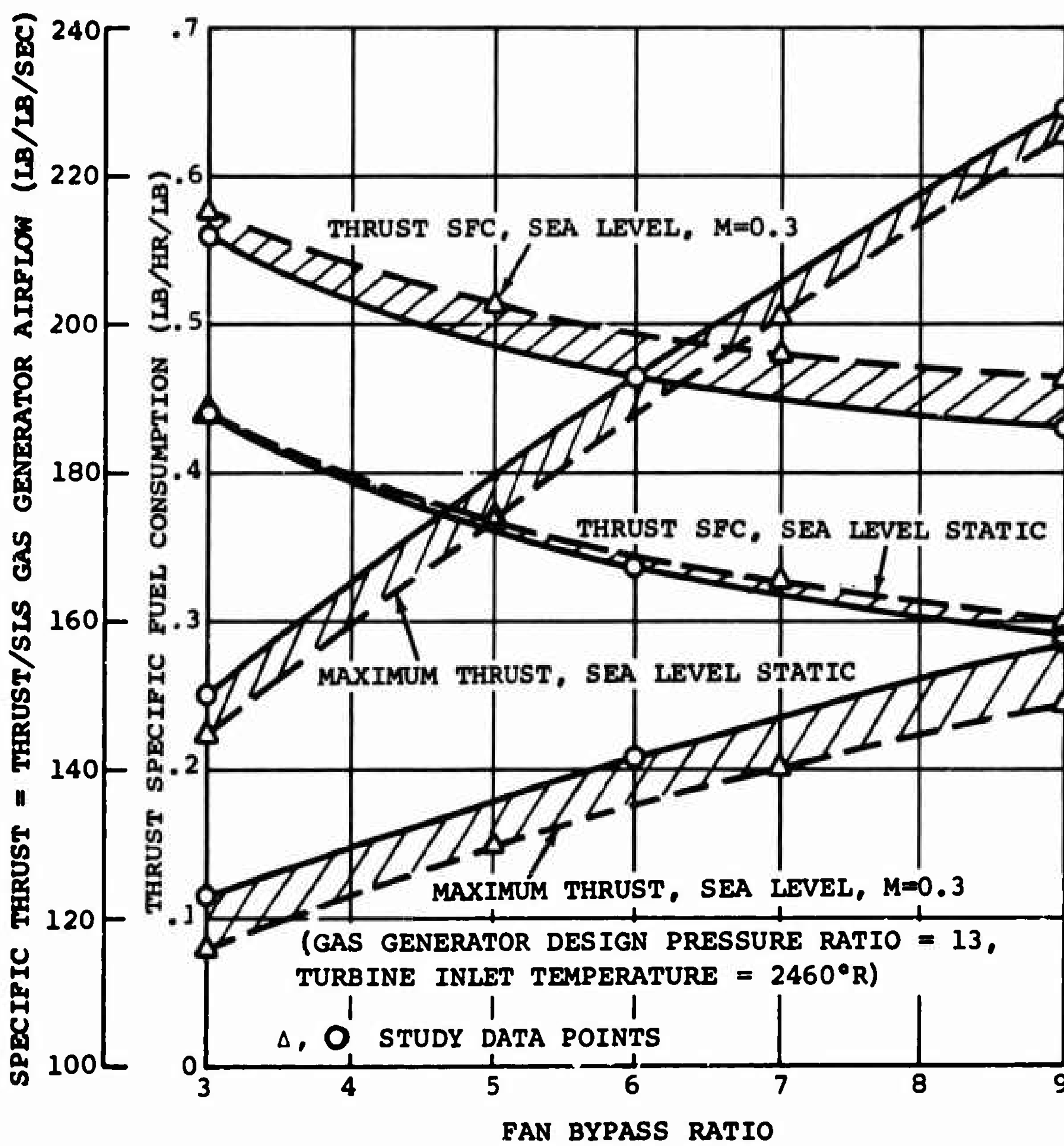


Figure 1. Typical Variation in Performance of Concentric Fans

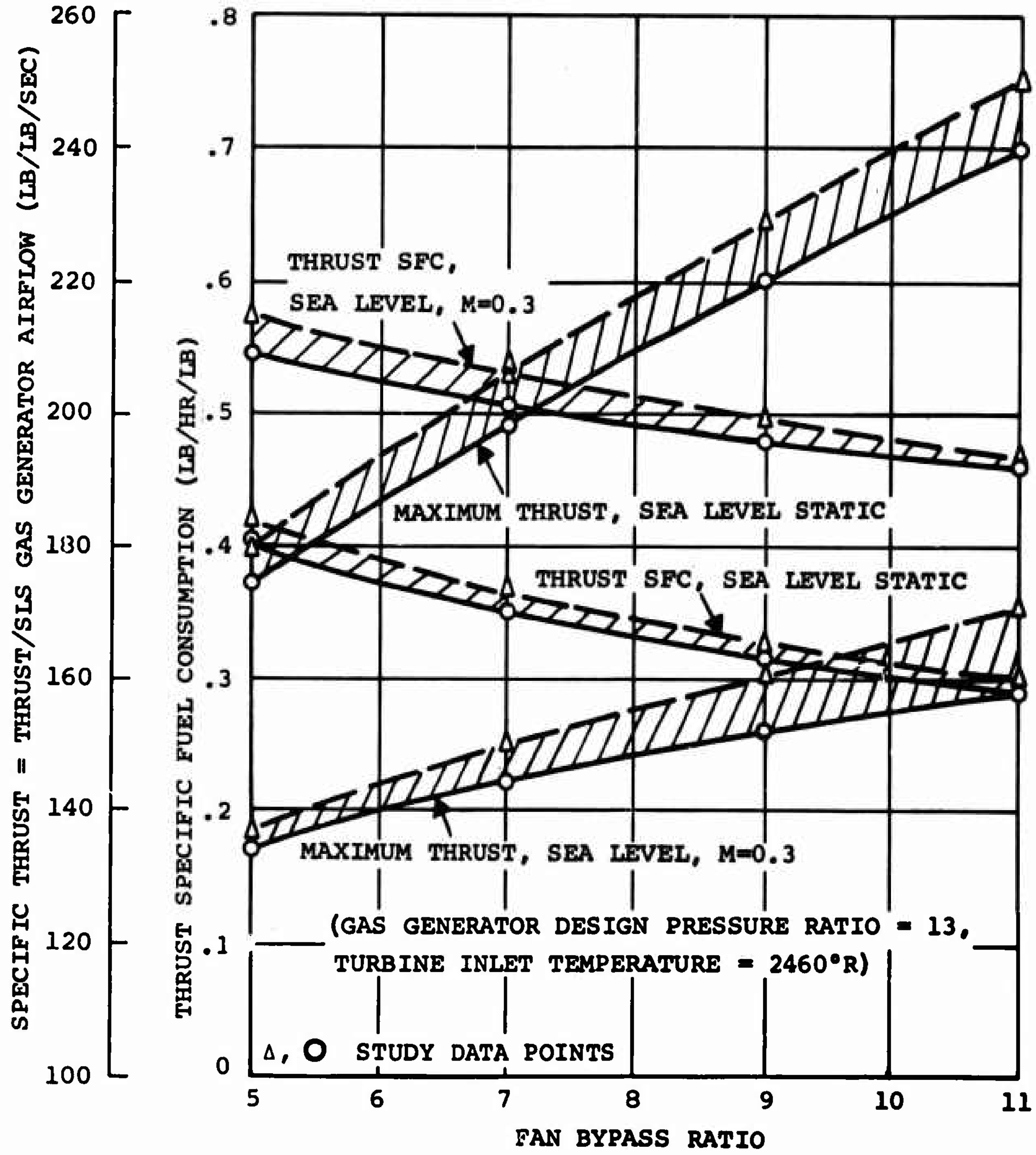


Figure 2. Typical Variation in Performance of Remote Gas-Coupled Fans

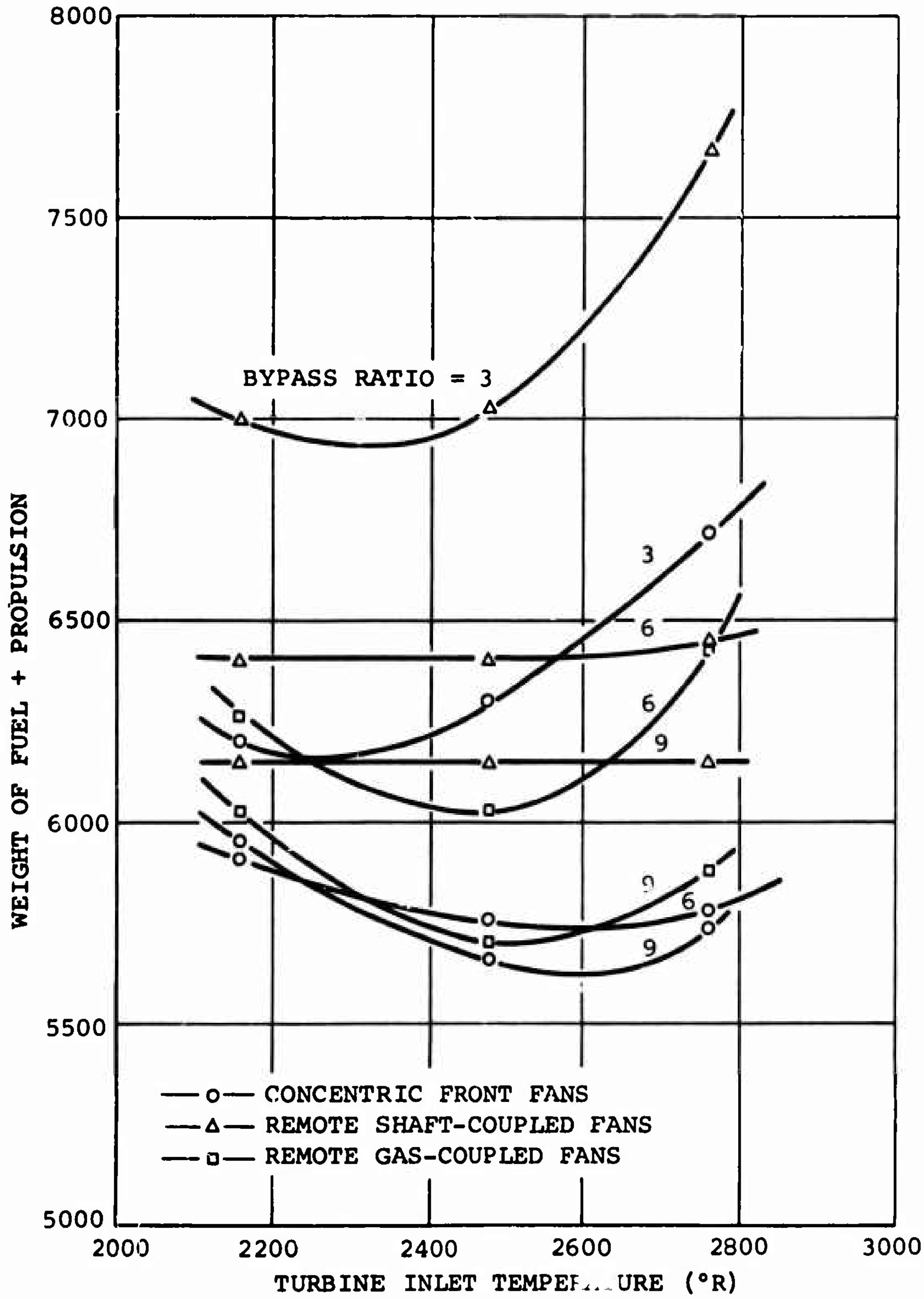


Figure 3. Fuel Plus Propulsion System Weight for Compound Helicopter Mission; Reference 1

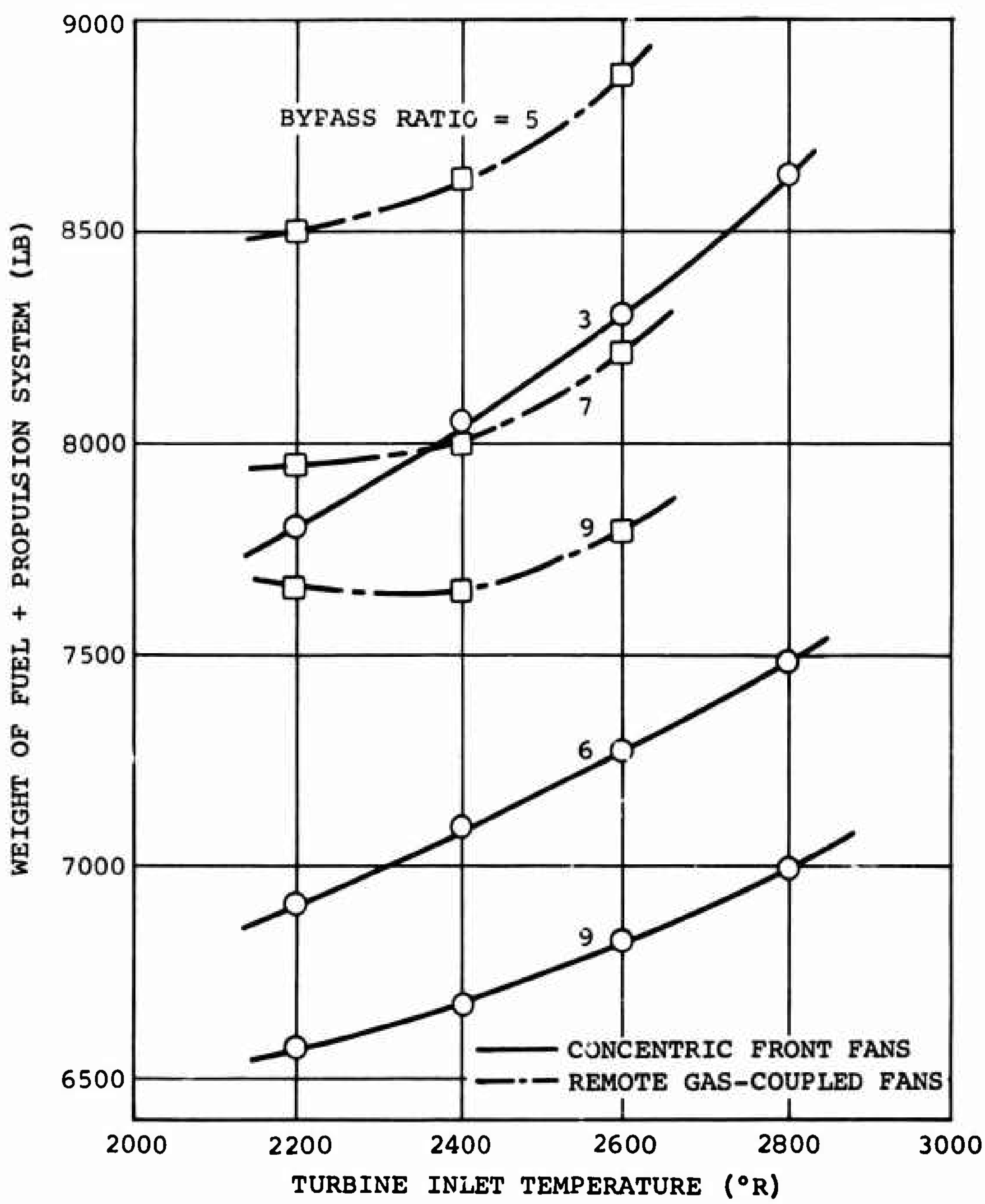


Figure 4. Fuel Plus Propulsion System Weight for Compound Helicopter Mission Using Data from Reference 7 and 27

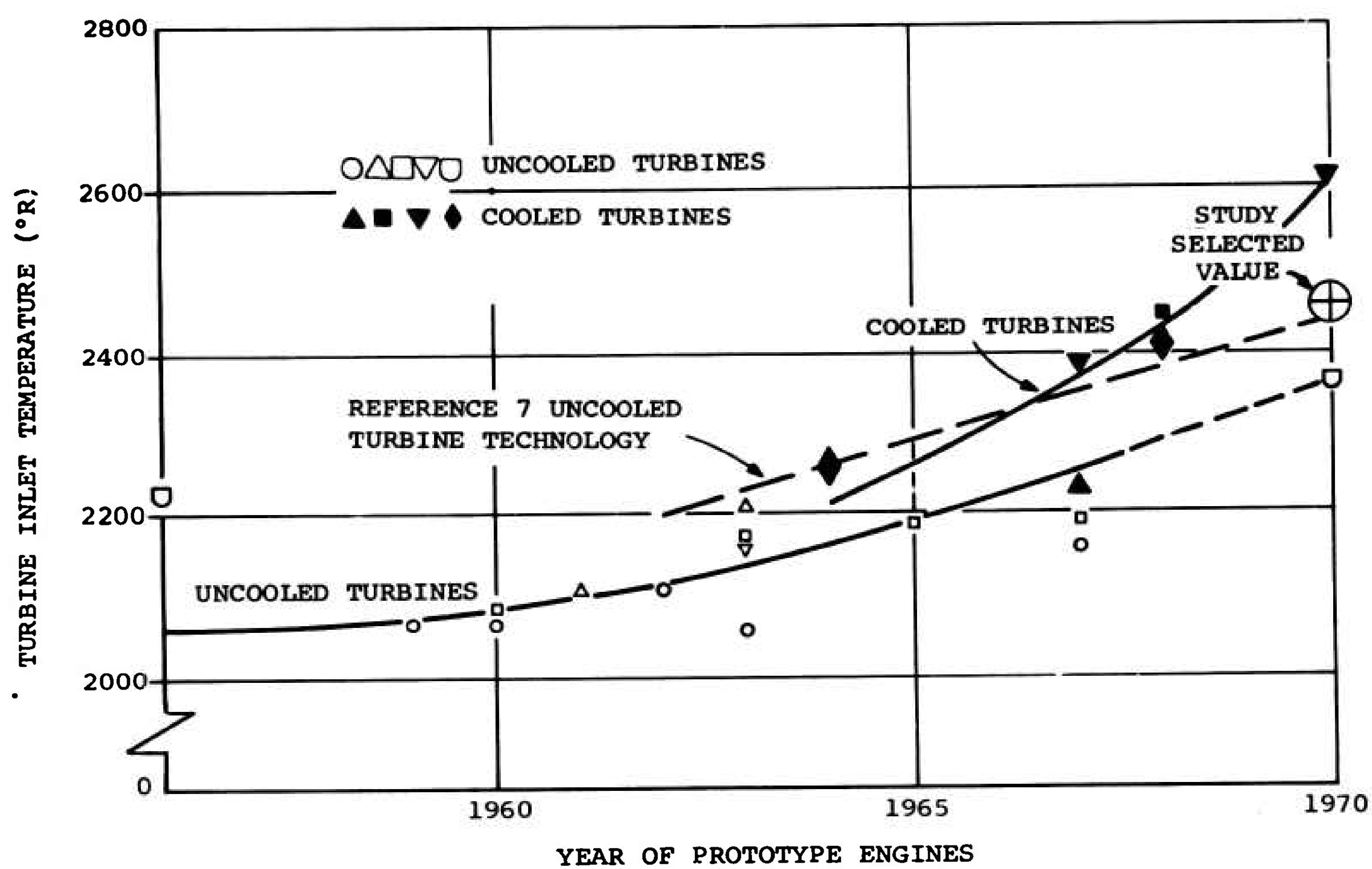


Figure 5. State-Of-The-Art Turbine Inlet Temperature (Military Rating)

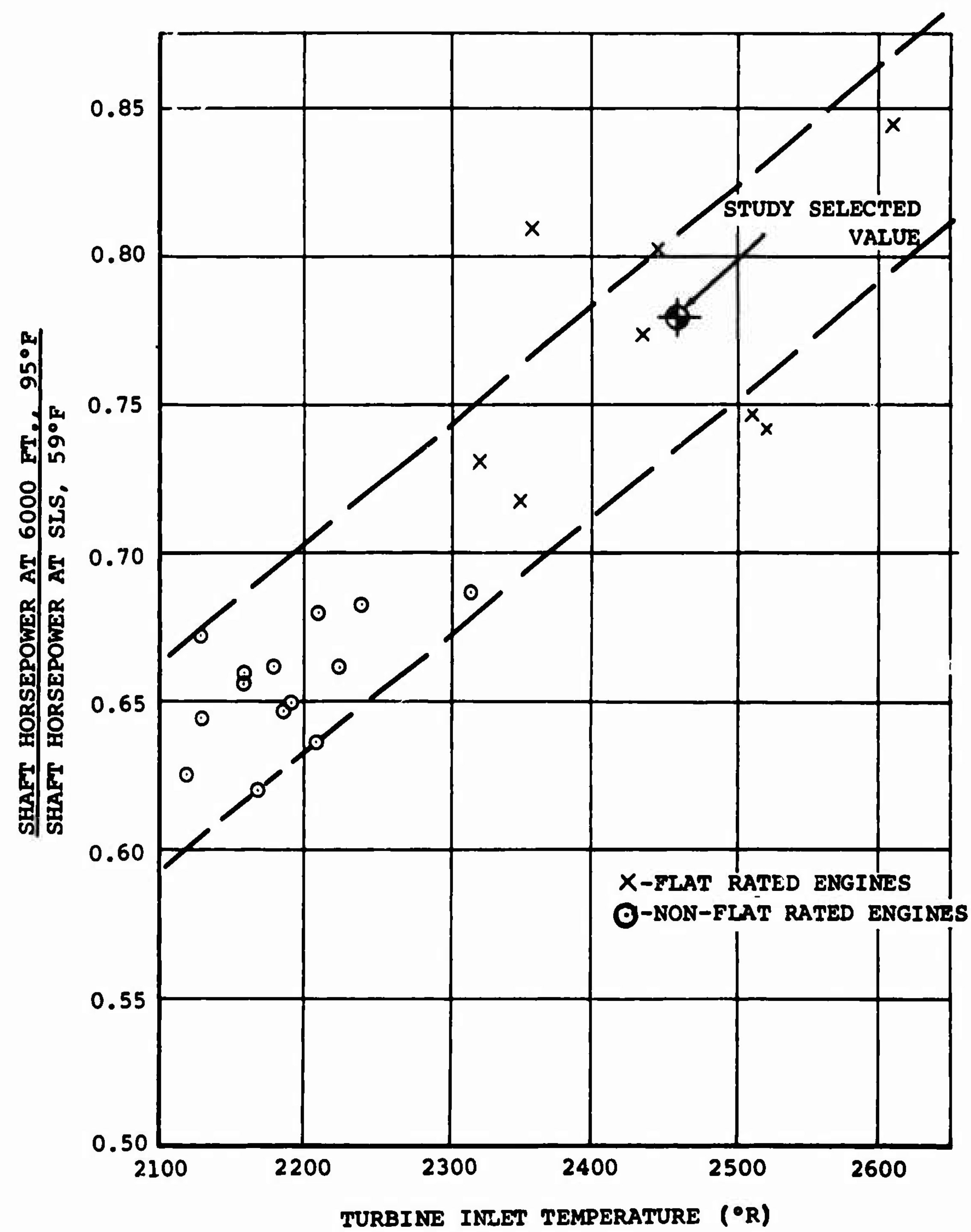
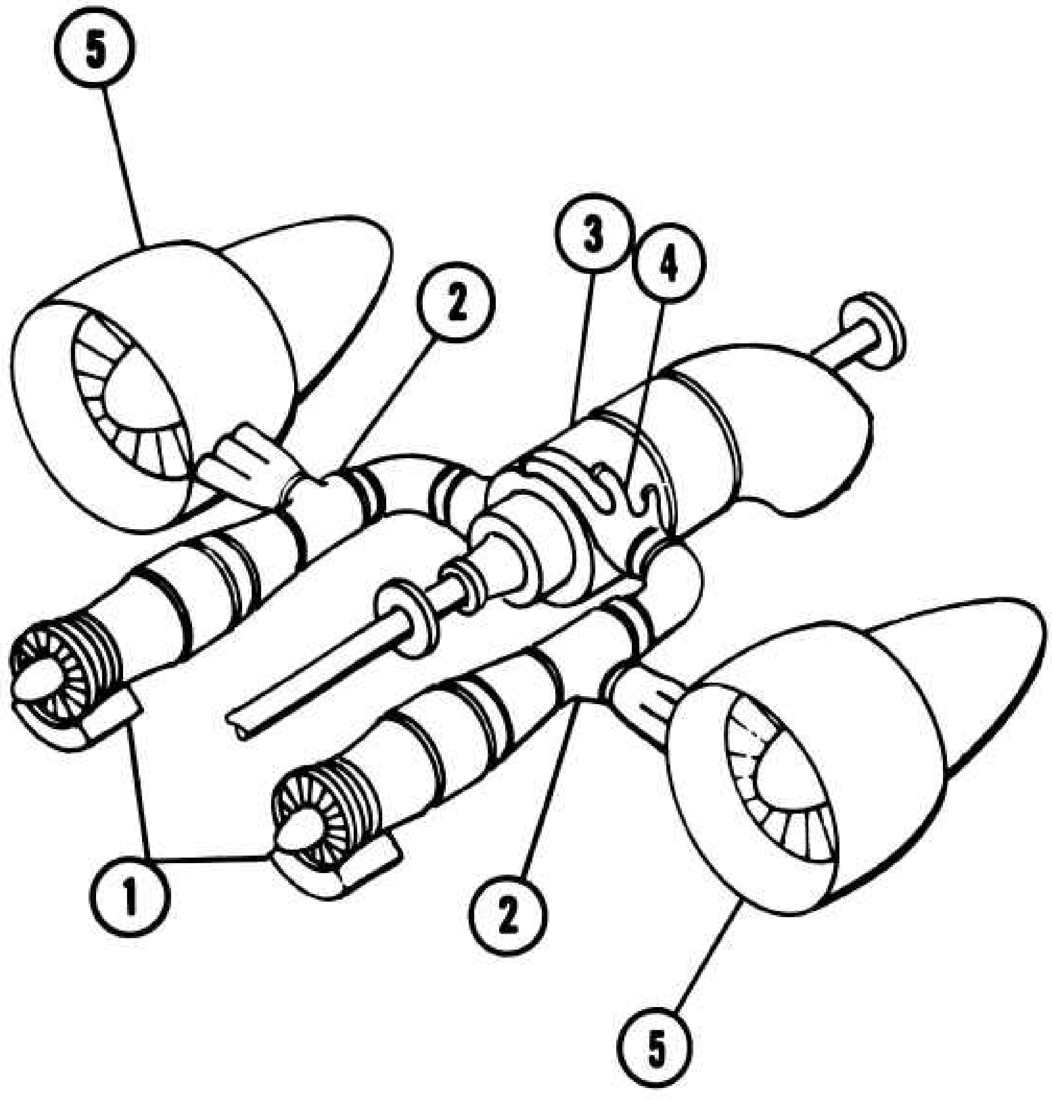
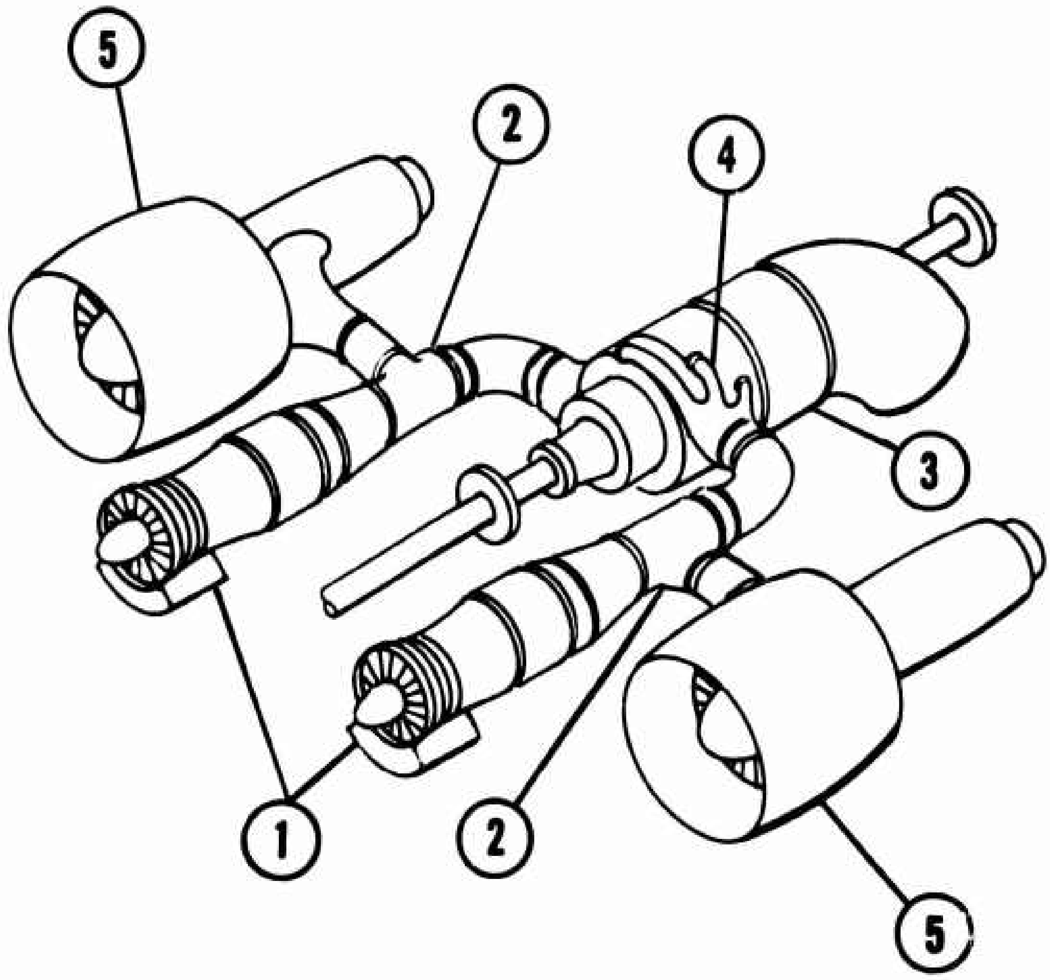


Figure 6. Turboshaft Engine Lapse Rate Data



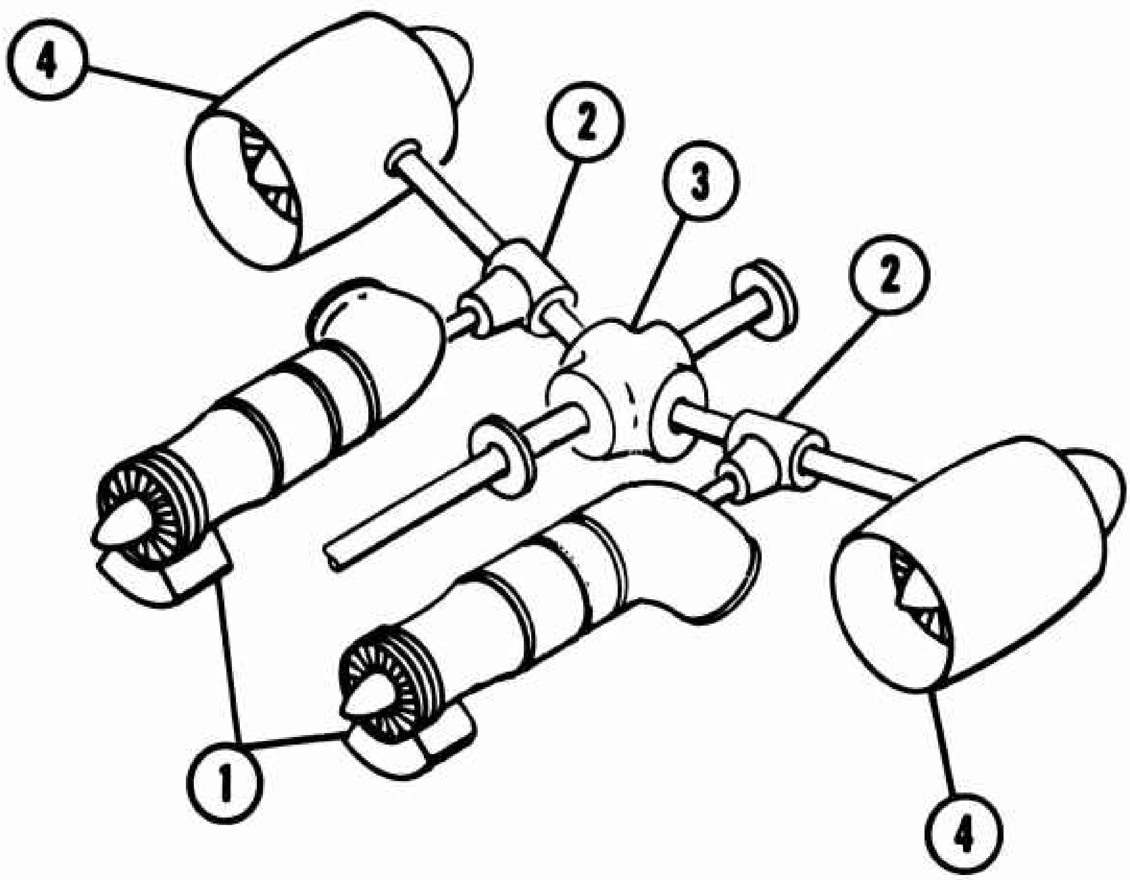
1 GAS GENERATOR
2 DIVERTER VALVE
3 REMOTE POWER TURBINE
4 VARIABLE ADMISSION ARCHES
5 TIP-TURBINE CRUISE FAN

Figure 7. Tip-Turbine Fan Propulsion System Conceptual Design (System 1a)



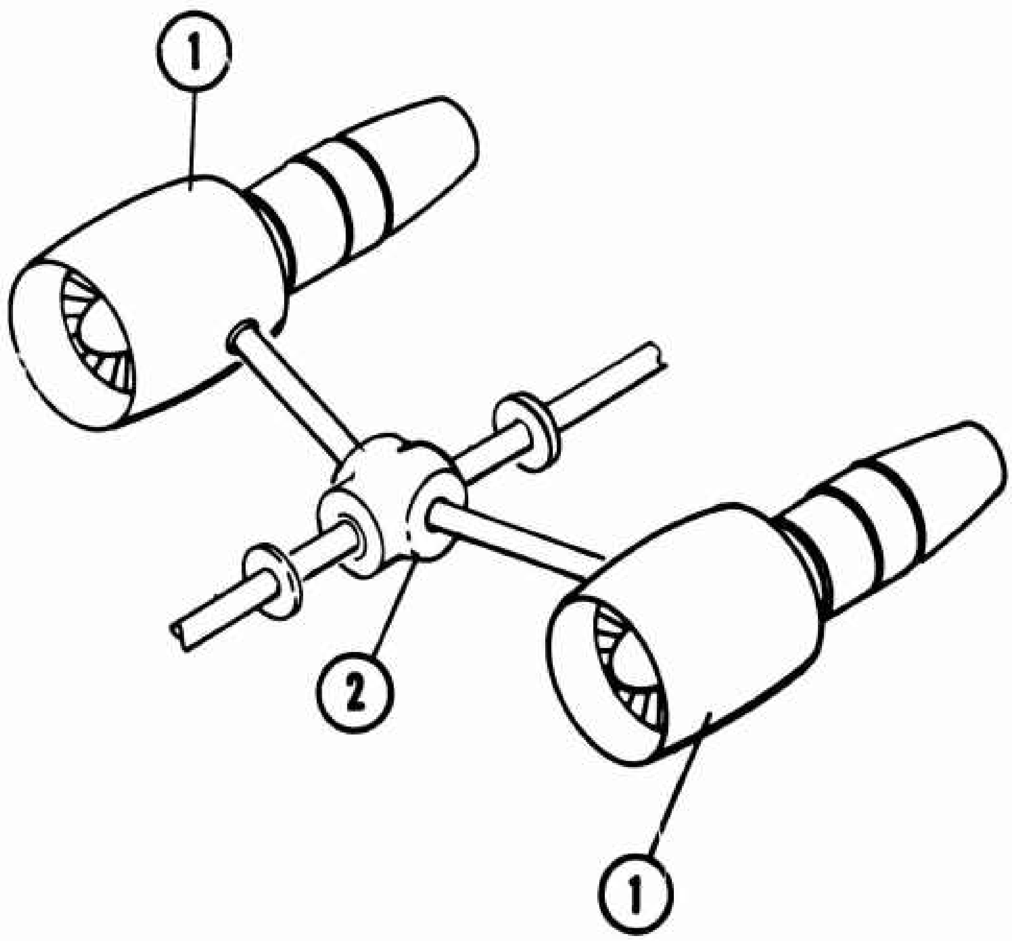
1 GAS GENERATOR
2 DIVERTER VALVE
3 REMOTE POWER TURBINE
4 VARIABLE ADMISSION ARCHES
5 AXIAL-TURBINE CRUISE FAN

Figure 8. Axial-Turbine Fan Propulsion System Conceptual Design (System 1b)



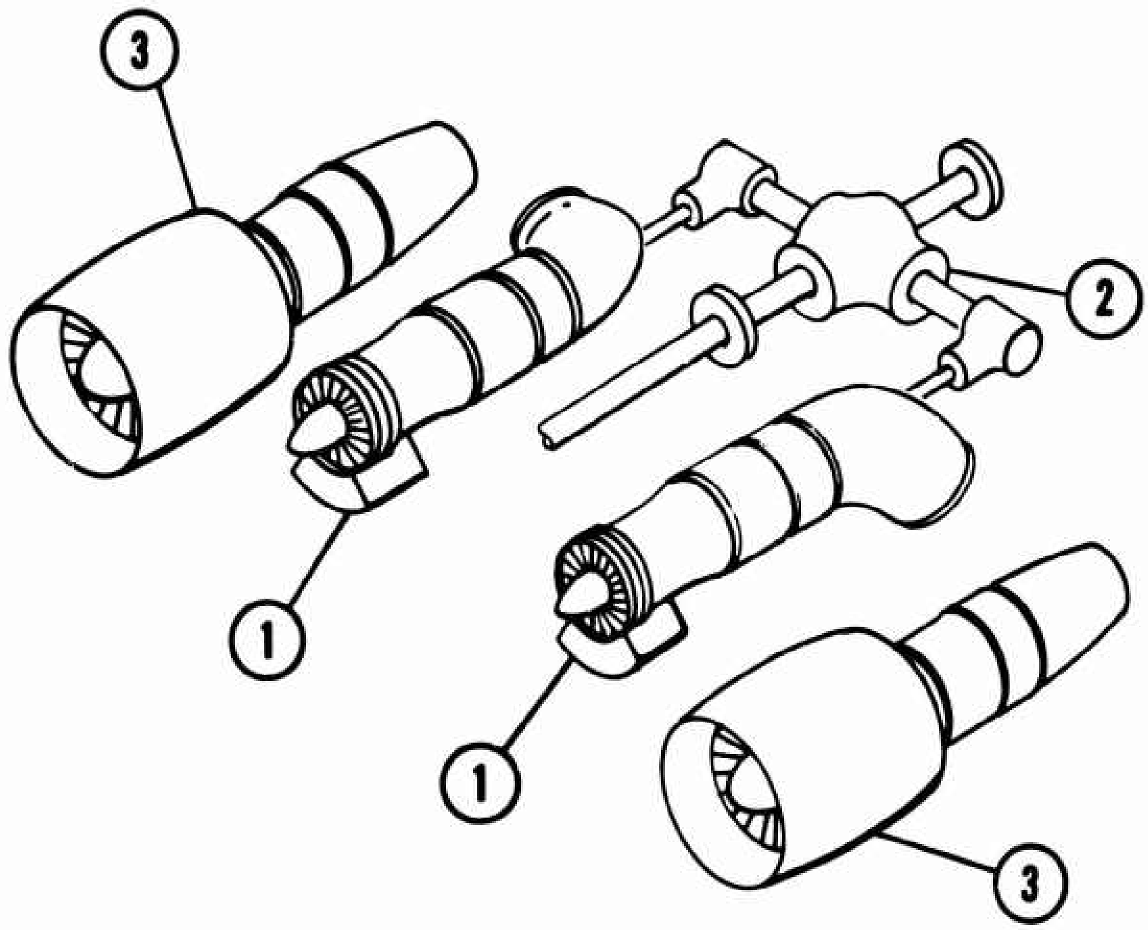
1 TURBOSHAFT ENGINE
2 GEARBOX
3 ROTOR SYSTEM COUPLING GEARBOX
4 REMOTE SHAFT-COUPLED FAN

Figure 9. Remote Shaft-Coupled Fan Propulsion System Conceptual Design (System 2a)



1 CONVERTIBLE CONCENTRIC FRONT FAN
2 ROTOR SYSTEM COUPLING GEARBOX

Figure 10. Convertible Concentric Front-Fan Propulsion System Conceptual Design (System 2b)



1 TURBOSHAFT ENGINE
2 ROTOR SYSTEM COUPLING GEARBOX
3 INDEPENDENT CONCENTRIC CRUISE FAN

Figure 11. Independent Propulsion System Conceptual Design (System 3)

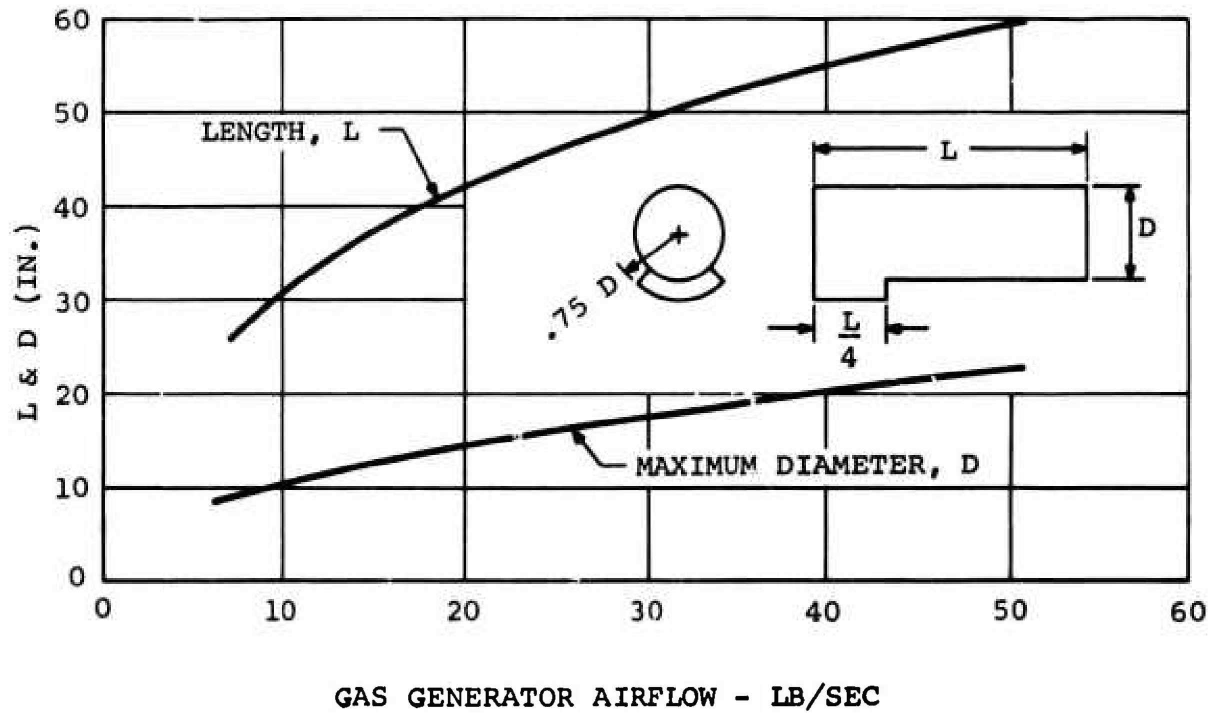


Figure 12. Gas Generator Dimensions for Remote Gas-Coupled Fan Propulsion System (Systems 1a and 1b)

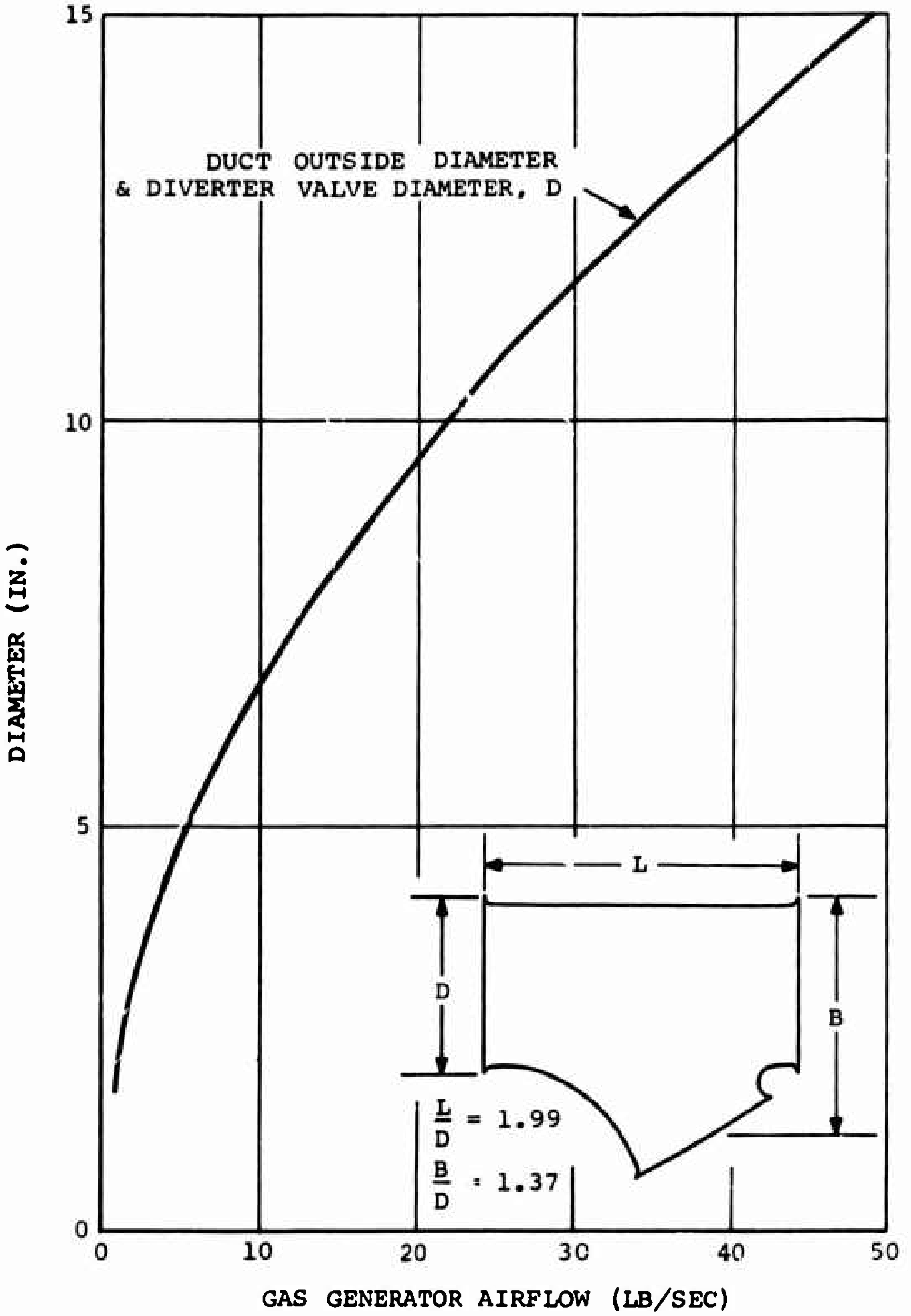


Figure 13. Diverter Valve and Duct Dimensions (Systems 1a and 1b)

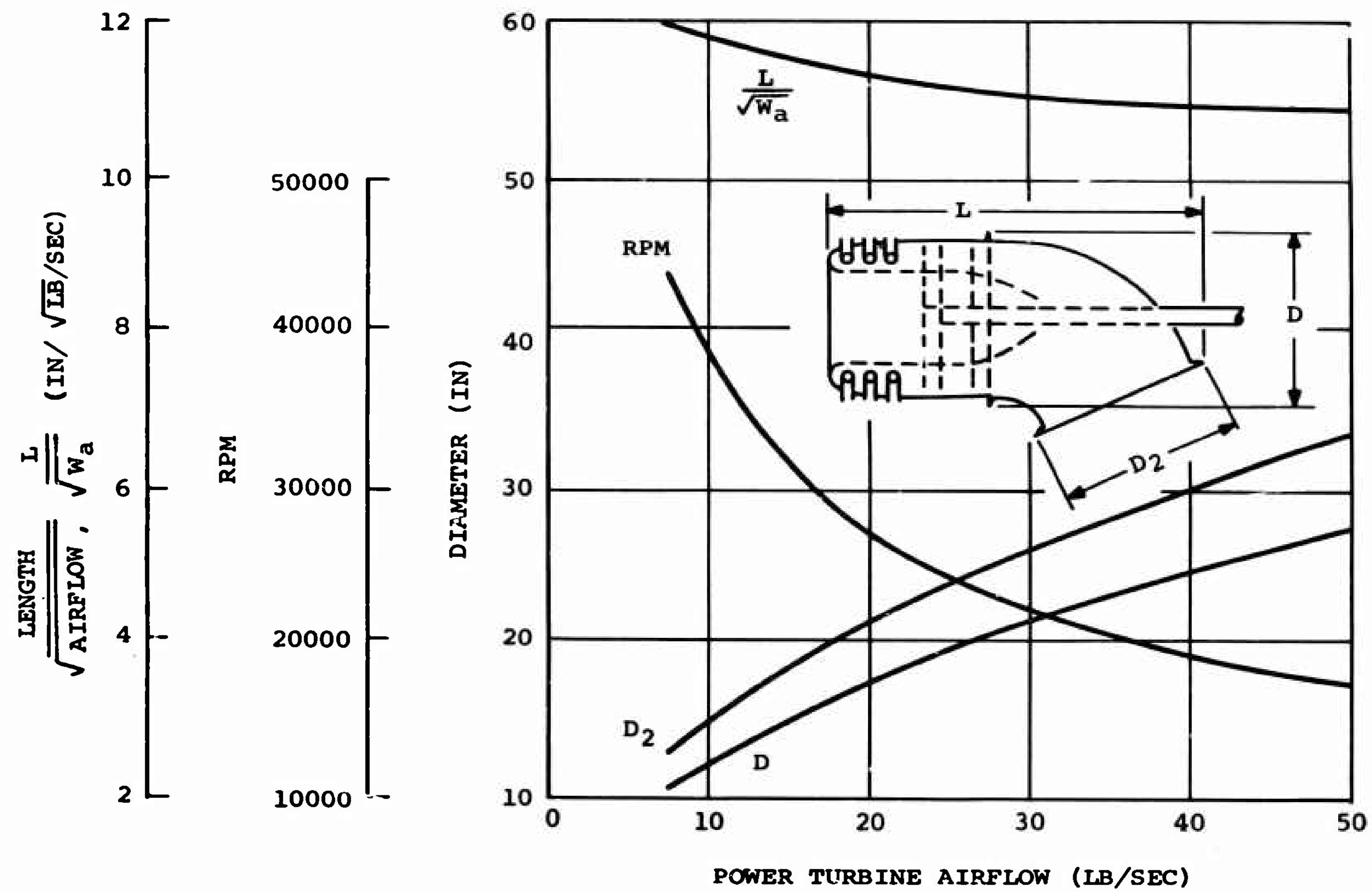
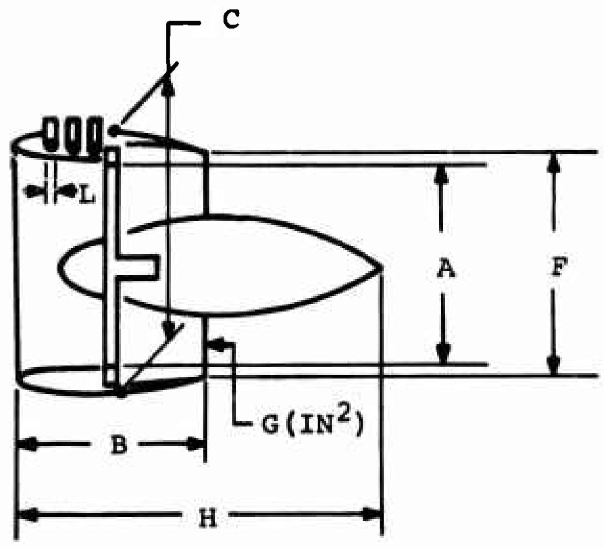


Figure 14. Remote Power Turbine Dimensions (No Gearbox) (Systems 1a and 1b)

(W_a = Fan Tip-Turbine Airflow, lb/sec)



BYPASS RATIO	$\frac{A}{\sqrt{W_a}}$	$\frac{C}{\sqrt{W_a}}$	$\frac{G}{W_a}$
3	4.75	6.55	8.50
6	6.12	7.80	17.60
9	7.10	9.15	27.00
12	8.05	9.80	36.50

$\frac{B}{\sqrt{W_a}} = 5.24$

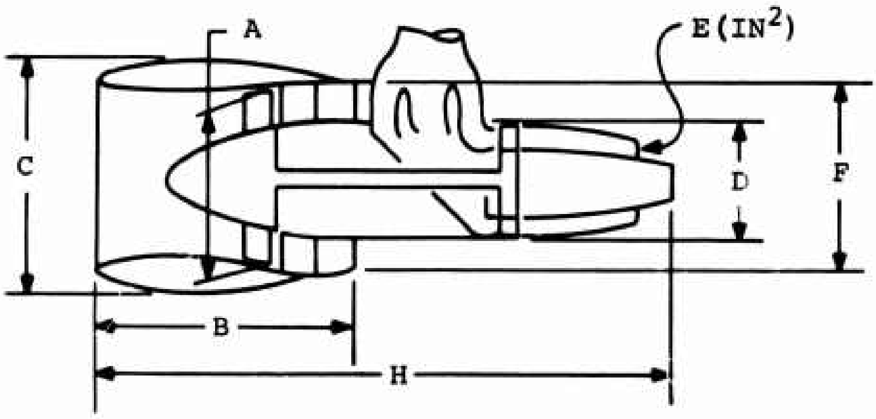
$\frac{F}{C} = 0.90$

$\frac{L}{\sqrt{W_a}} = 0.640$

$H = B + 1.3F$

Figure 15. Remote Gas-Coupled Tip-Turbine Fan Dimensions (System 1a)

(W_a = Fan-Turbine Airflow, lb/sec)



Bypass Ratio	$\frac{A}{\sqrt{W_a}}$	$\frac{D}{\sqrt{W_a}}$	$\frac{E}{W_a}$
3	6.40	3.75	4.90
6	6.90	3.75	4.65
9	7.65	3.75	4.70
12	8.50	3.75	4.85

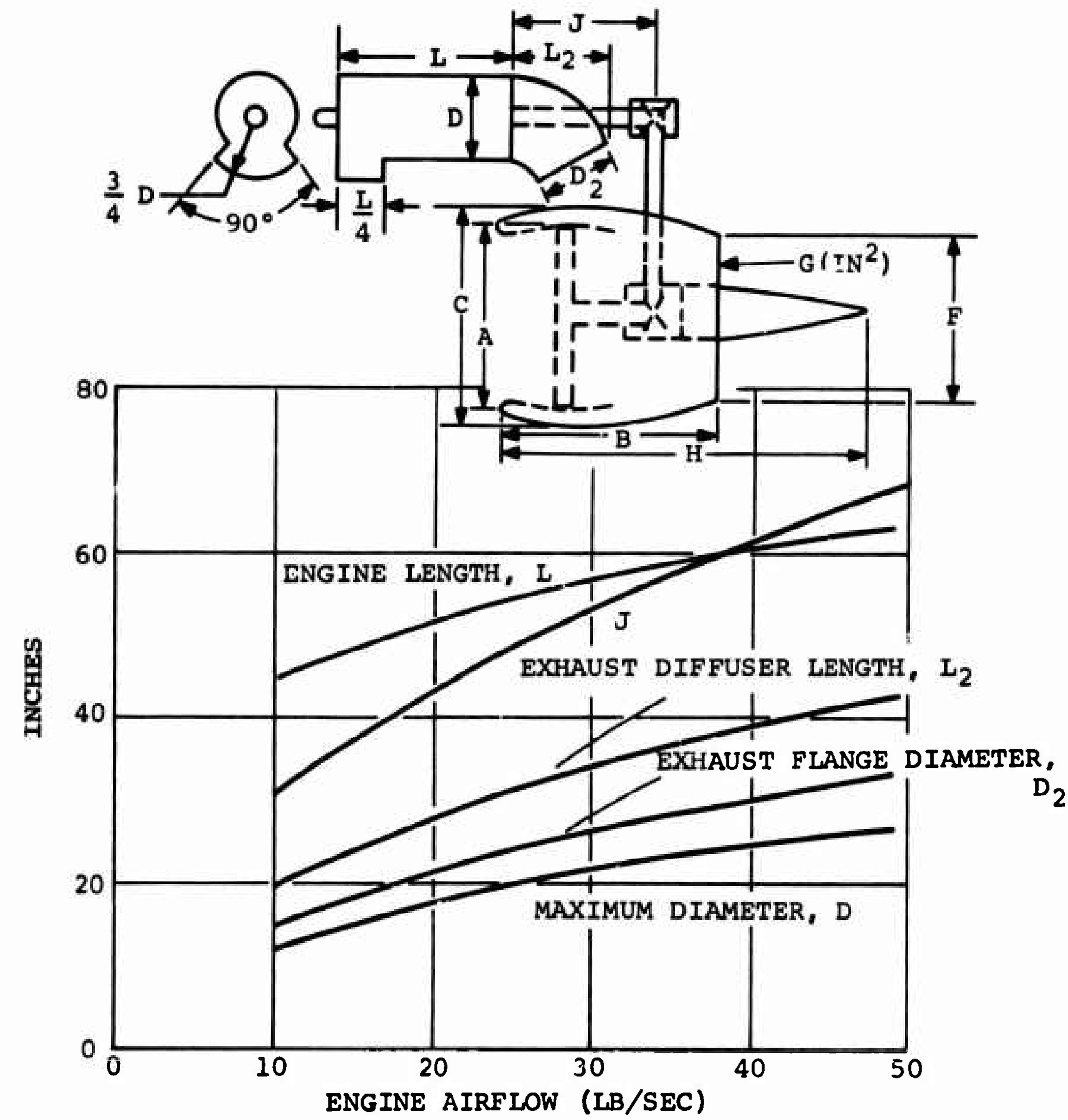
$\frac{B}{A} = 1.074$

$\frac{C}{A} = 1.04$, $(C-A)_{minimum} \approx 3.0$ IN.

$\frac{F}{A} = 0.987$

$\frac{H}{A} = 2.39$

Figure 16. Remote Gas-Coupled Axial-Turbine Fan Dimensions (System 1b)



BYPASS RATIO	$\frac{A}{\sqrt{W_a'}}$	$\frac{G}{W_a'}$
3	4.30	4.8
6	5.60	13.3
9	6.85	21.9
12	8.10	30.4

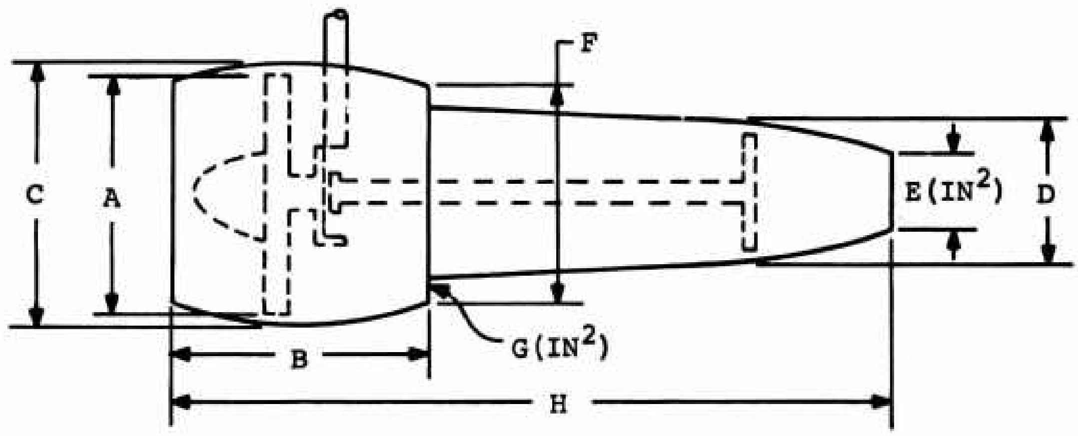
$\frac{B}{A} = 1.32;\ \frac{C}{A}\ 1.04,\ \frac{F}{C} = 0.90;\ \frac{H}{A} = 2.1$

$(C-A)_{MINIMUM} = 3.0$ IN.

W_a' = PART OF ENGINE AIRFLOW (LB/SEC) THAT PRODUCES FAN POWER

Figure 17. Remote Power Turbine Dimensions and RPM (No Gearbox) (Systems 1a and 1b)

(W_a = Gas Generator Airflow, lb/sec)



Bypass Ratio	$\frac{A}{\sqrt{W_a}}$	$\frac{D}{\sqrt{W_a}}$	$\frac{E}{W_a}$	$\frac{G}{W_a}$
3	5.32	3.72	4.90	5.5
6	6.12	3.72	4.65	13.0
9	7.16	3.72	4.70	23.5
12	8.35	3.72	4.85	36.0

$\frac{B}{A} = 1.20$

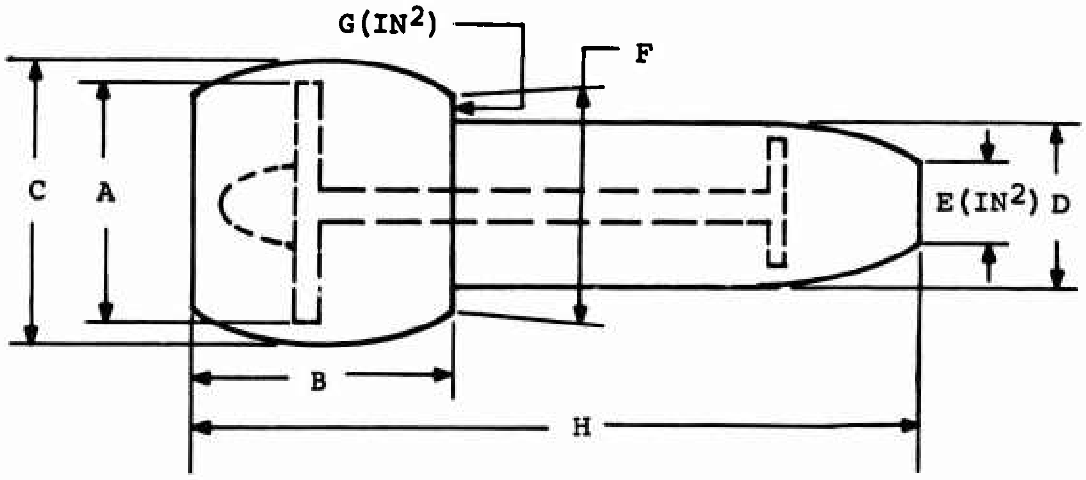
$\frac{C}{A} = 1.04 \quad (C-A)_{minimum} = 3.0$ IN.

$\frac{F}{A} = 0.936$

$\frac{H}{A} = 3.0$

Figure 18. Convertible Concentric Cruise-Fan Dimensions (System 2b)

(W_a = Gas Generator Airflow, lb/sec)



Bypass Ratio	$\frac{A}{\sqrt{W_a}}$	$\frac{D}{\sqrt{W_a}}$	$\frac{E}{W_a}$	$\frac{G}{W_a}$
3	5.32	4.10	4.90	5.5
6	6.12	4.20	4.65	13.0
9	7.16	4.42	4.70	23.5
12	8.35	4.95	4.85	36.0

$\frac{B}{A} = 1.20$

$\frac{C}{A} = 1.04 \quad (C-A)_{minimum} = 3.0$ IN.

$\frac{F}{A} = 0.936$

$\frac{H}{A} = 3.42$

Figure 19. Concentric Cruise Fan Dimensions (System 3)

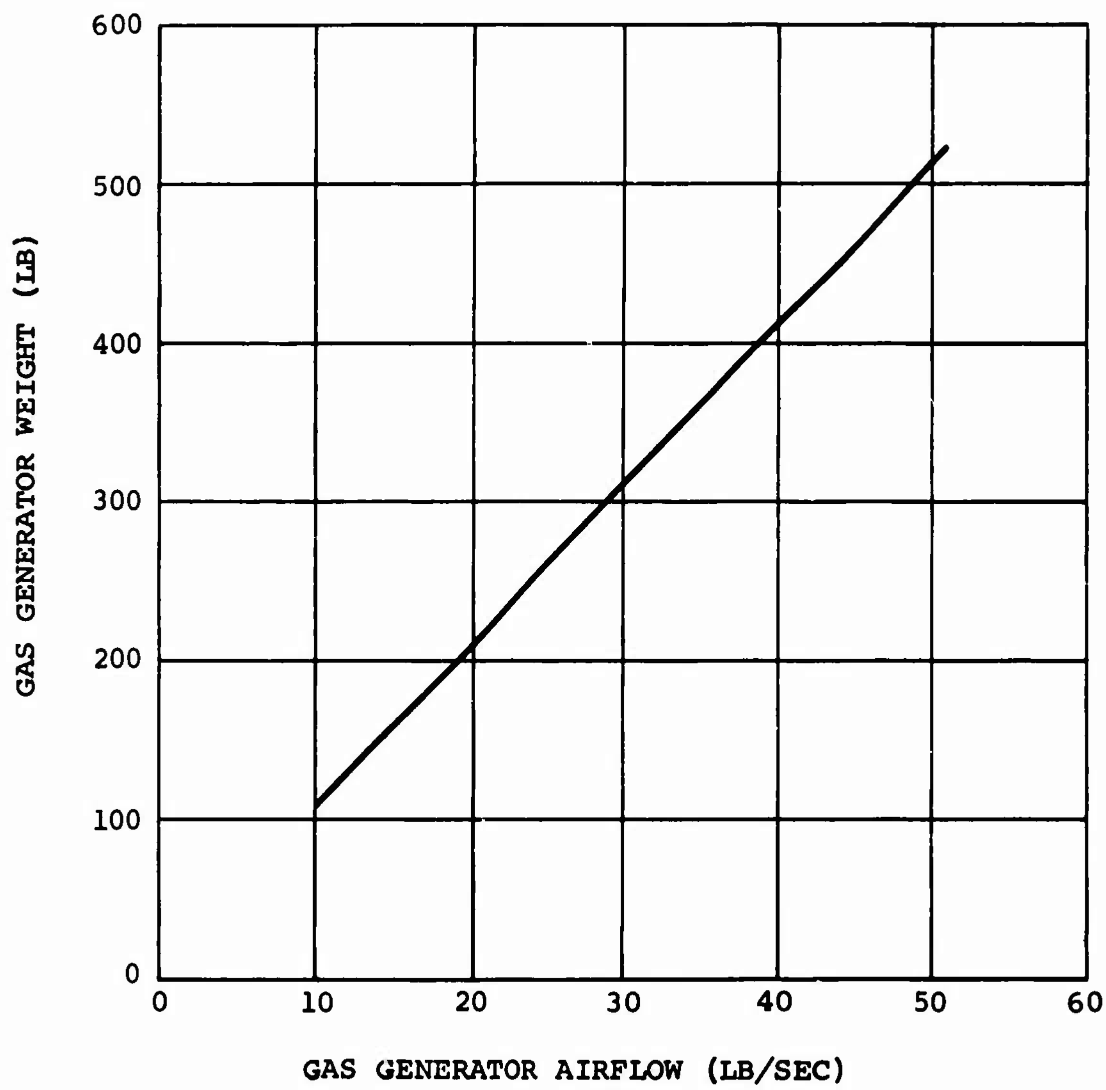


Figure 20. Weight of Gas Generator, Including Controls and Accessories, for Remote Gas-Coupled Fan Systems (Systems 1a and 1b)

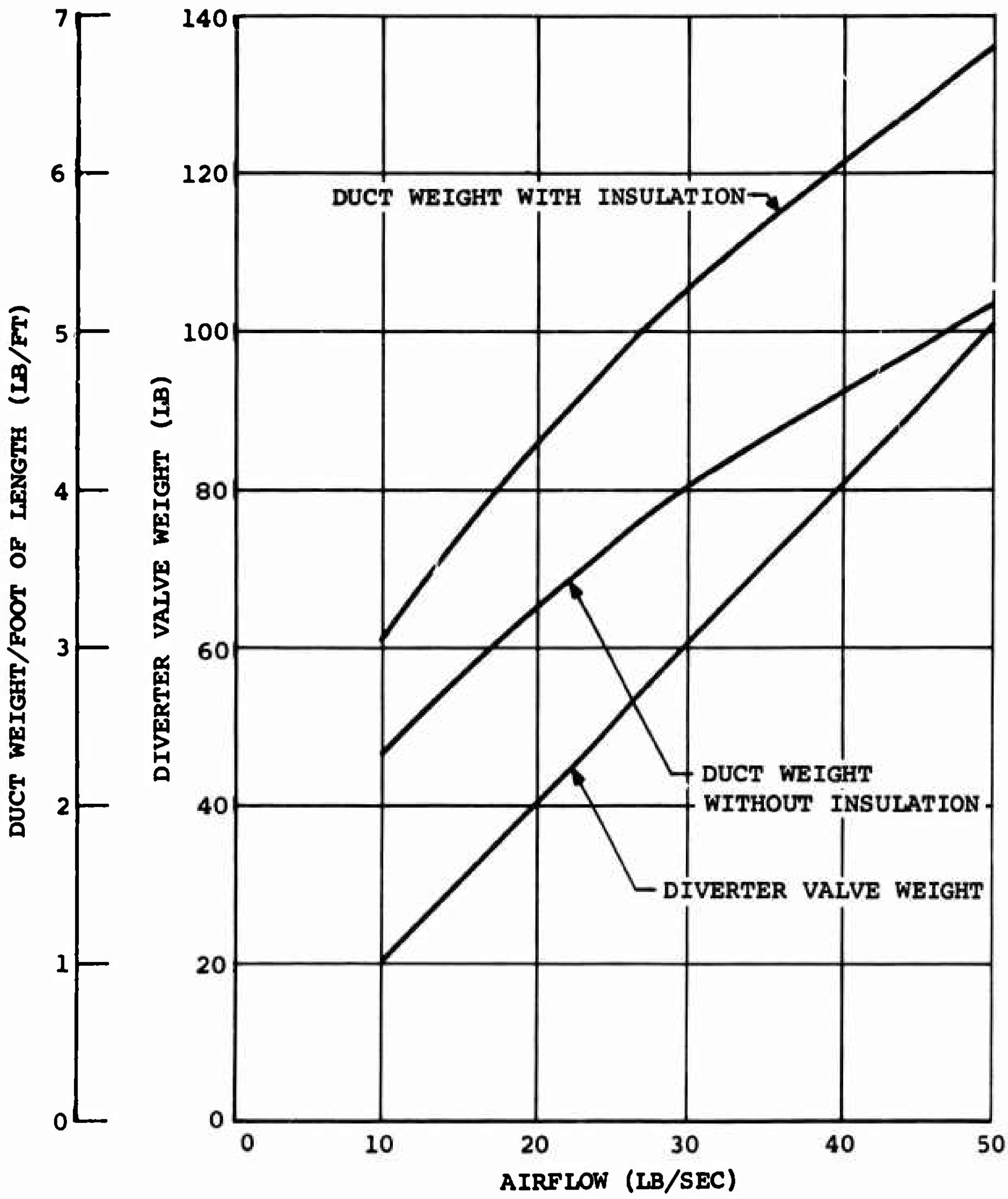


Figure 21. Diverter Valve and Duct Weights

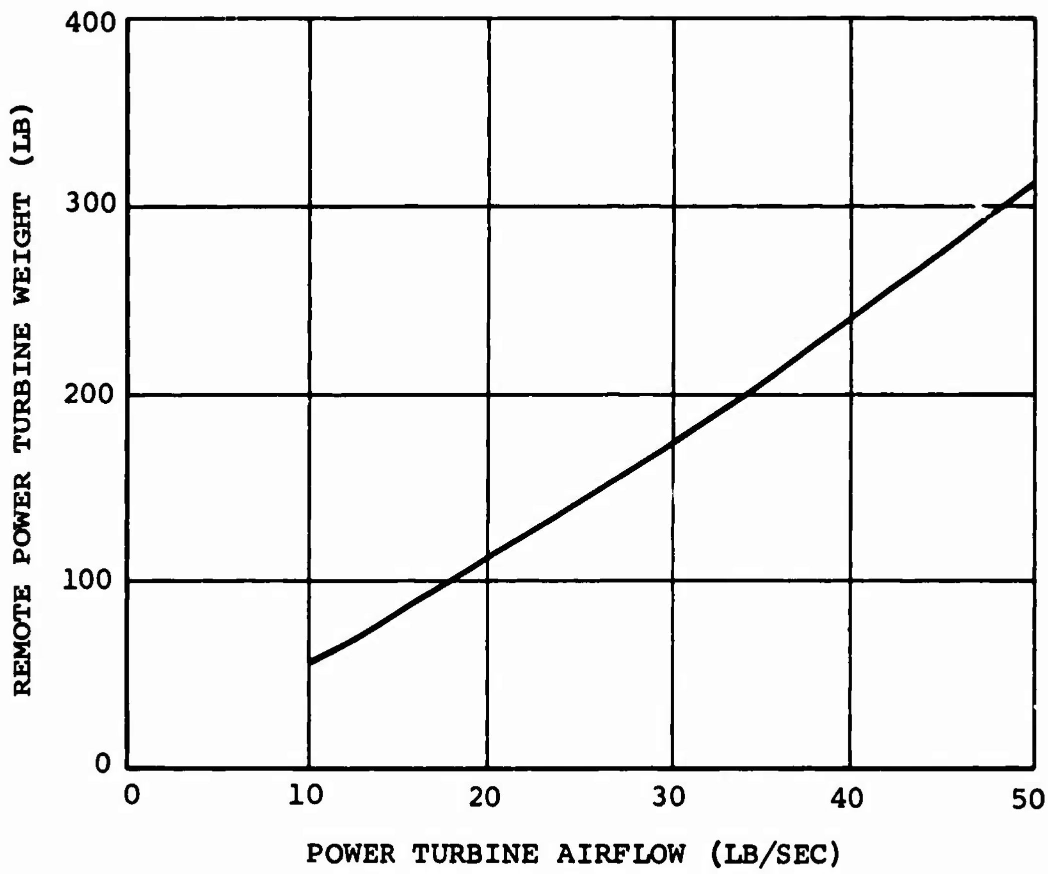


Figure 22. Remote Power Turbine Weight for Remote Gas-Coupled Fan Systems (Systems 1a and 1b)

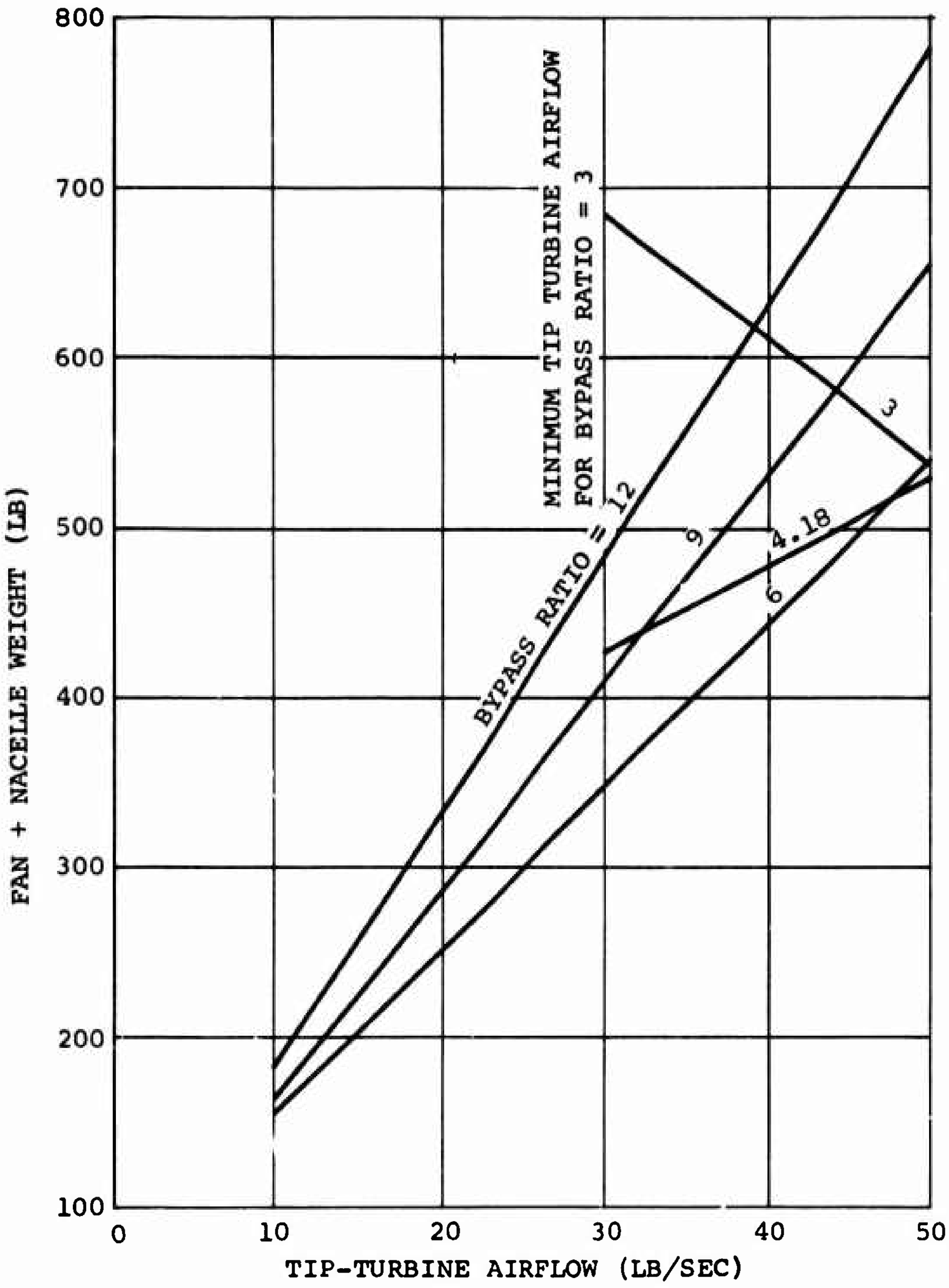


Figure 23. Weight of Fan and Nacelle for Remote Gas-Coupled Tip-Turbine Fan System (System 1a)

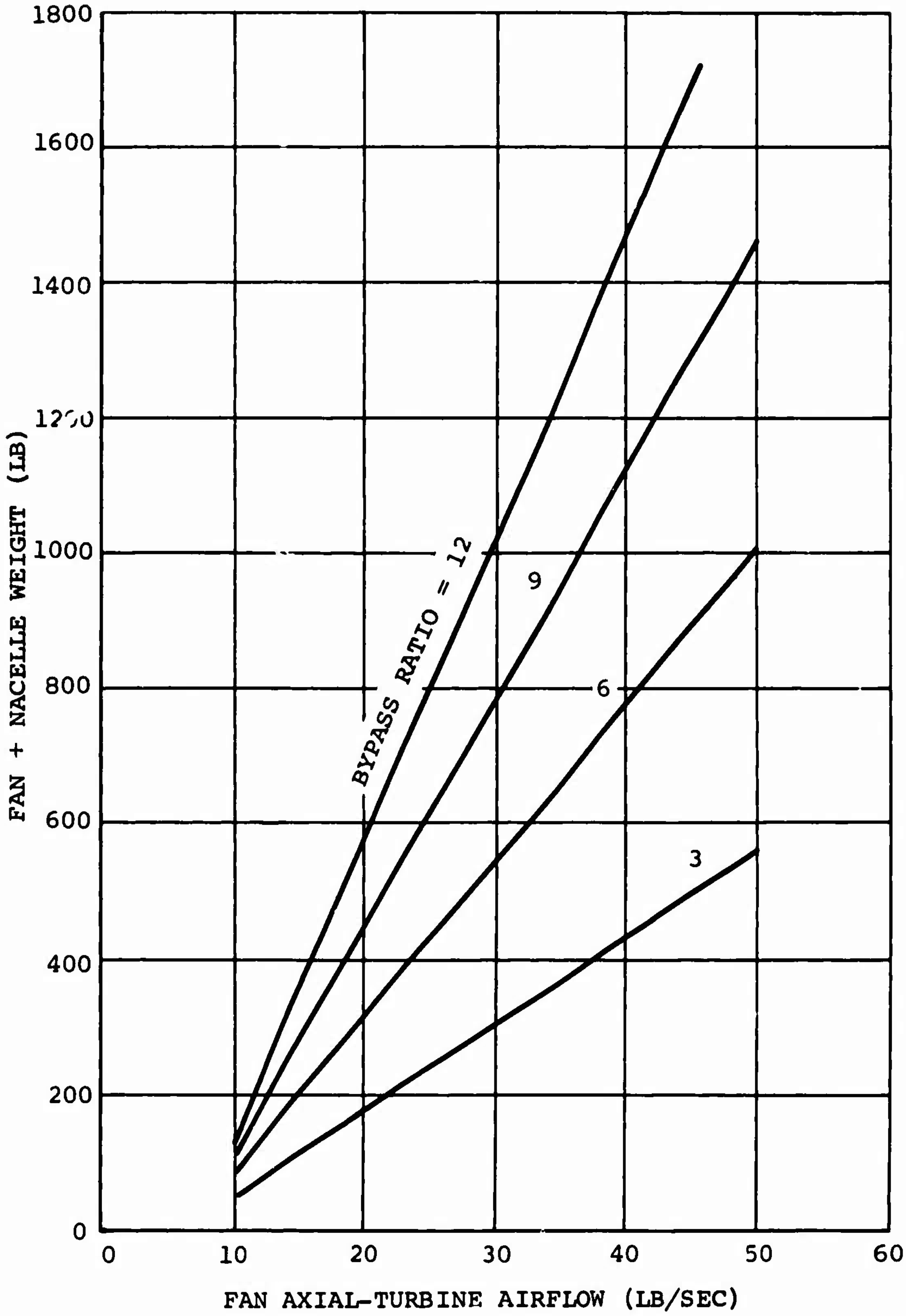


Figure 24. Weight of Fan and Nacelle for Remote Gas-Coupled Axial-Turbine Fan System (System 1b)

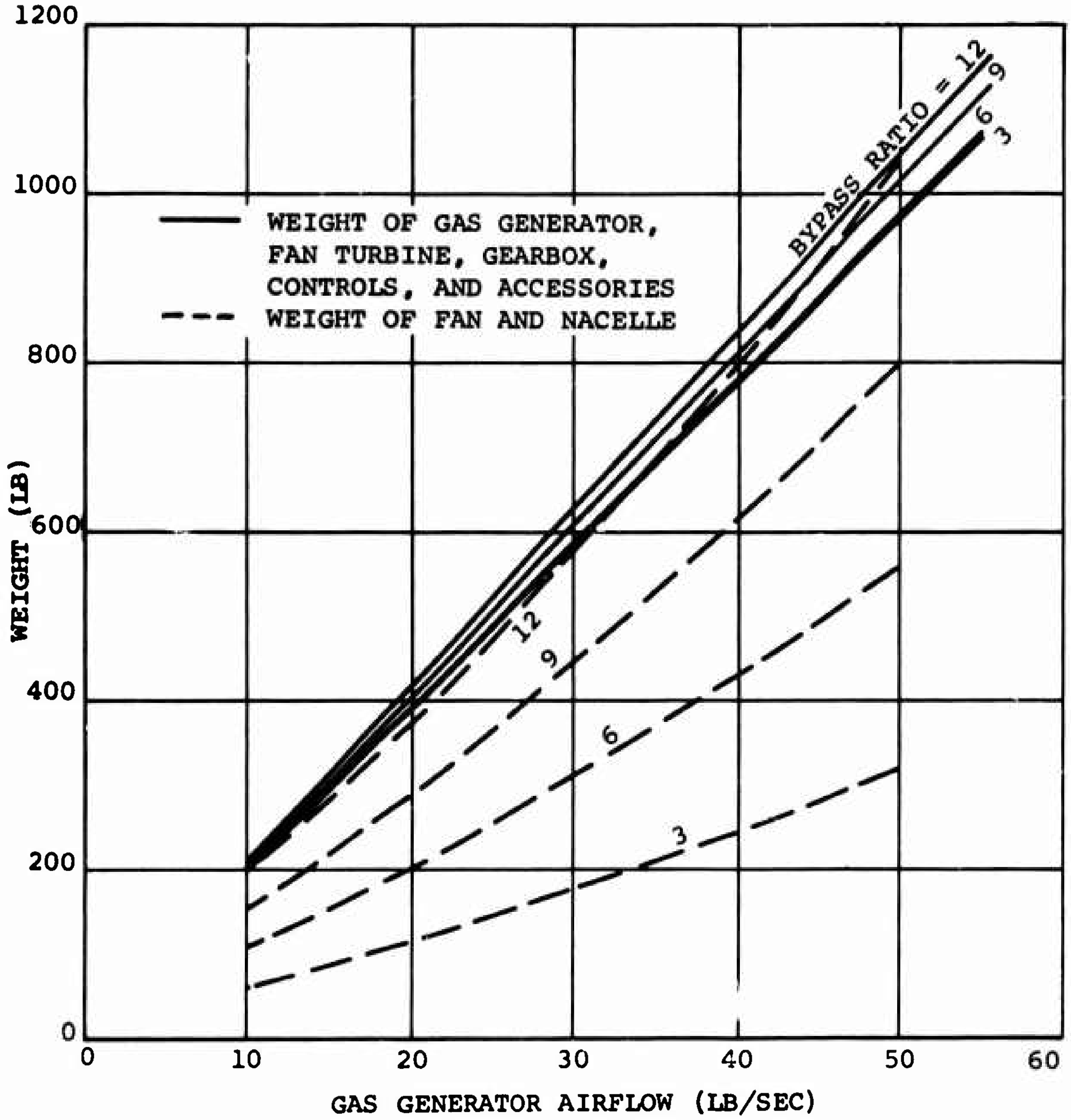


Figure 25. Weight of Remote Shaft-Coupled Fan System (System 2a)

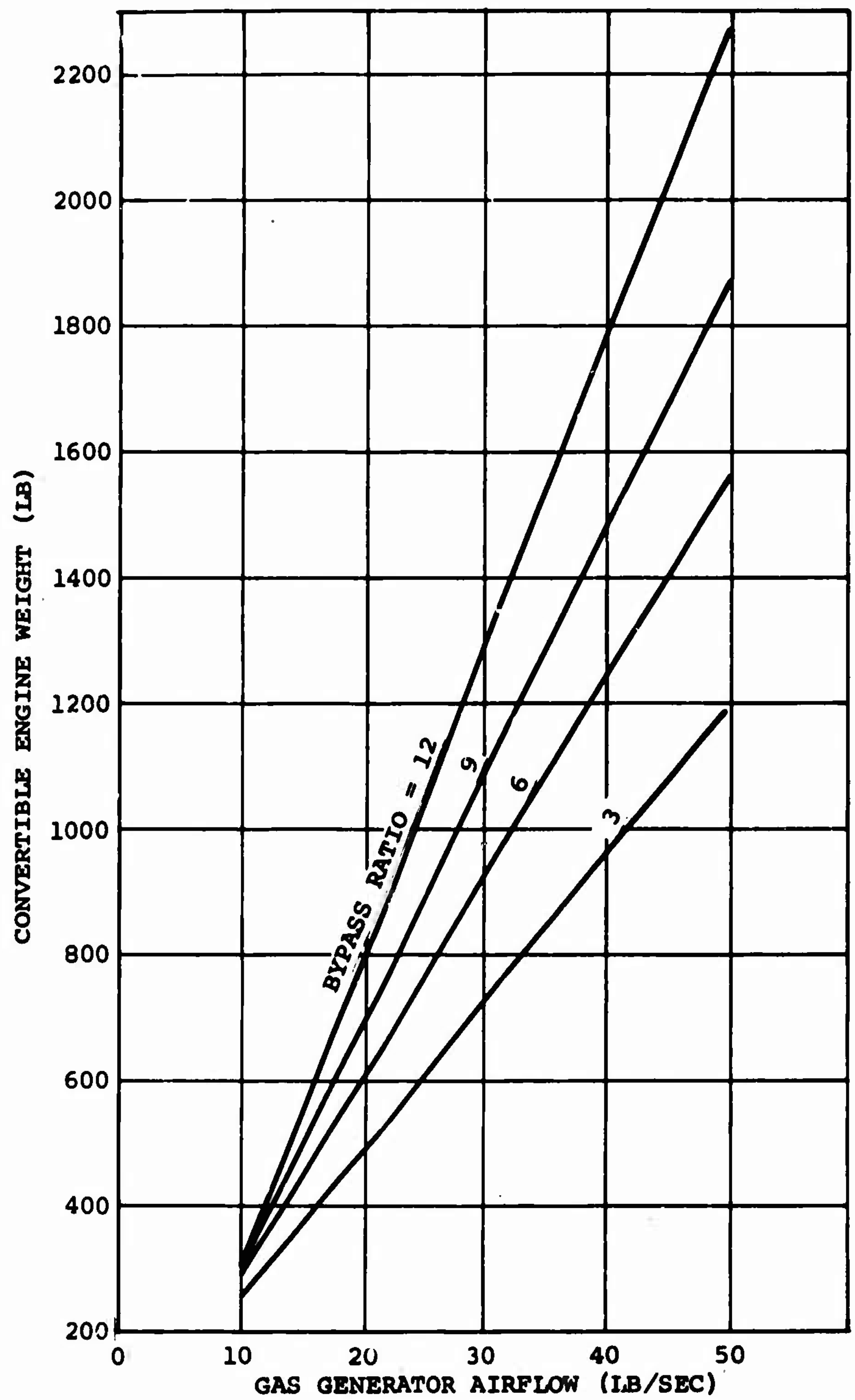


Figure 26. Weight of Convertible Cruise-Fan System (System 2b)

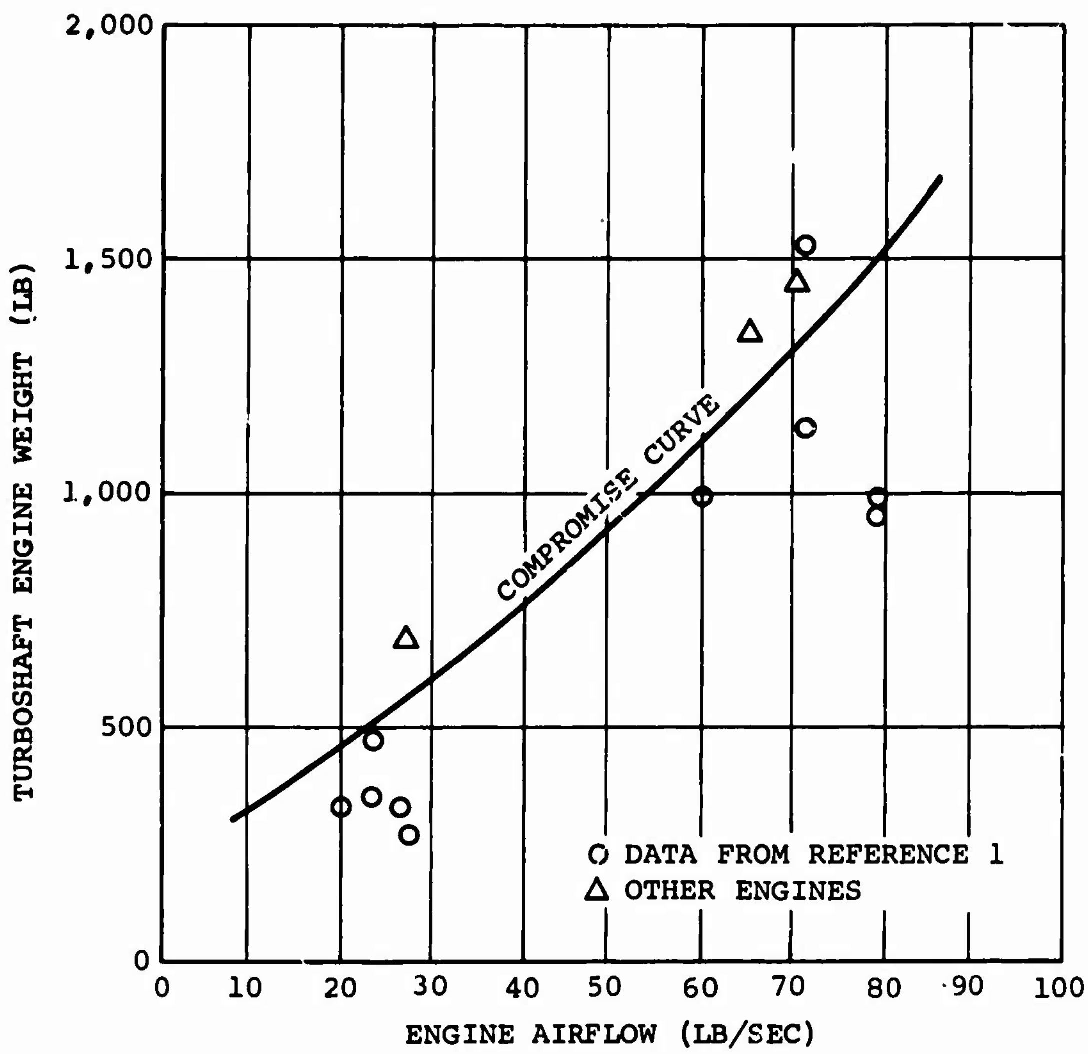


Figure 27. Turboshaft Engine Weight for Independent Concentric Front-Fan Propulsion System (System 3)

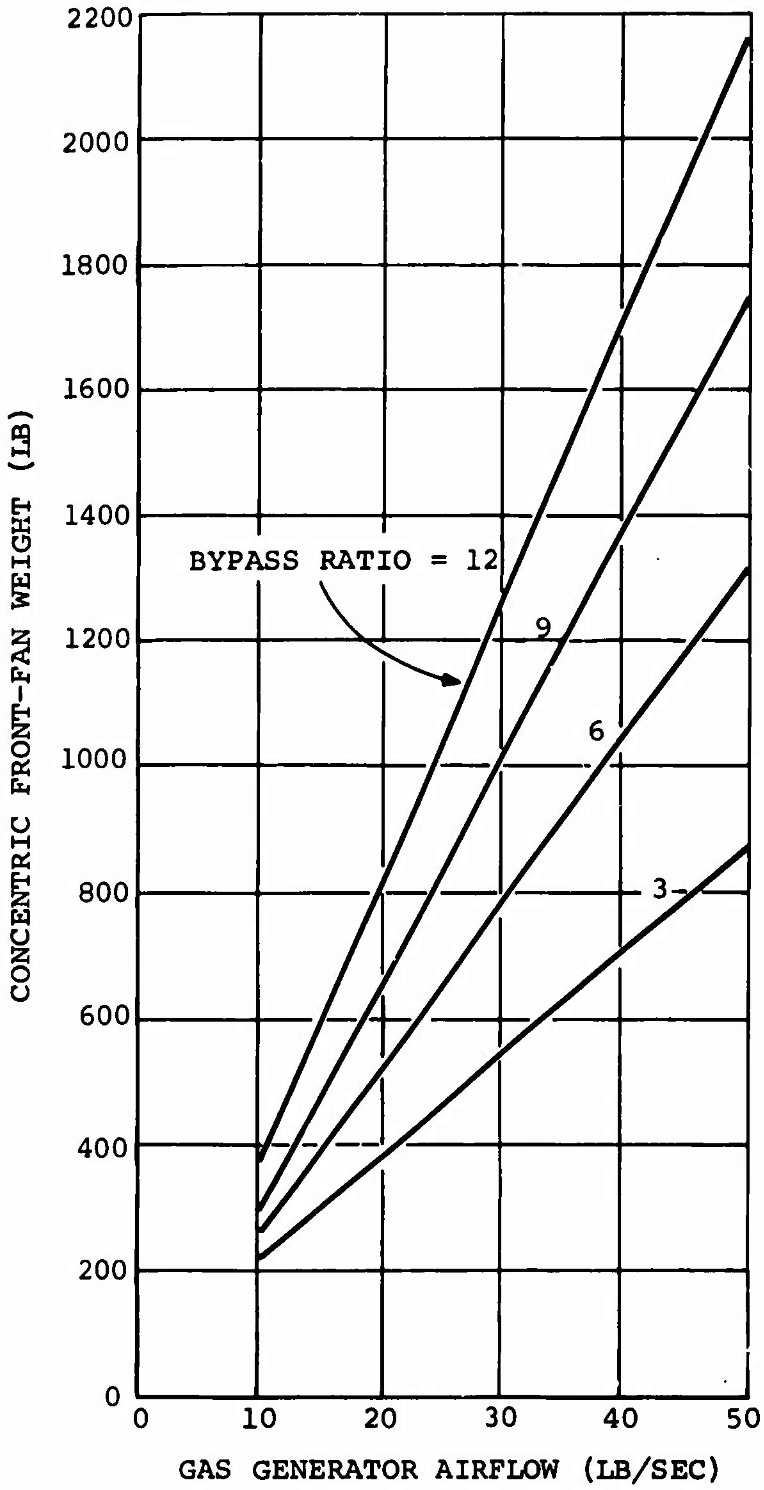


Figure 28. Weight of Independent Concentric Front-Fan System (System 3)

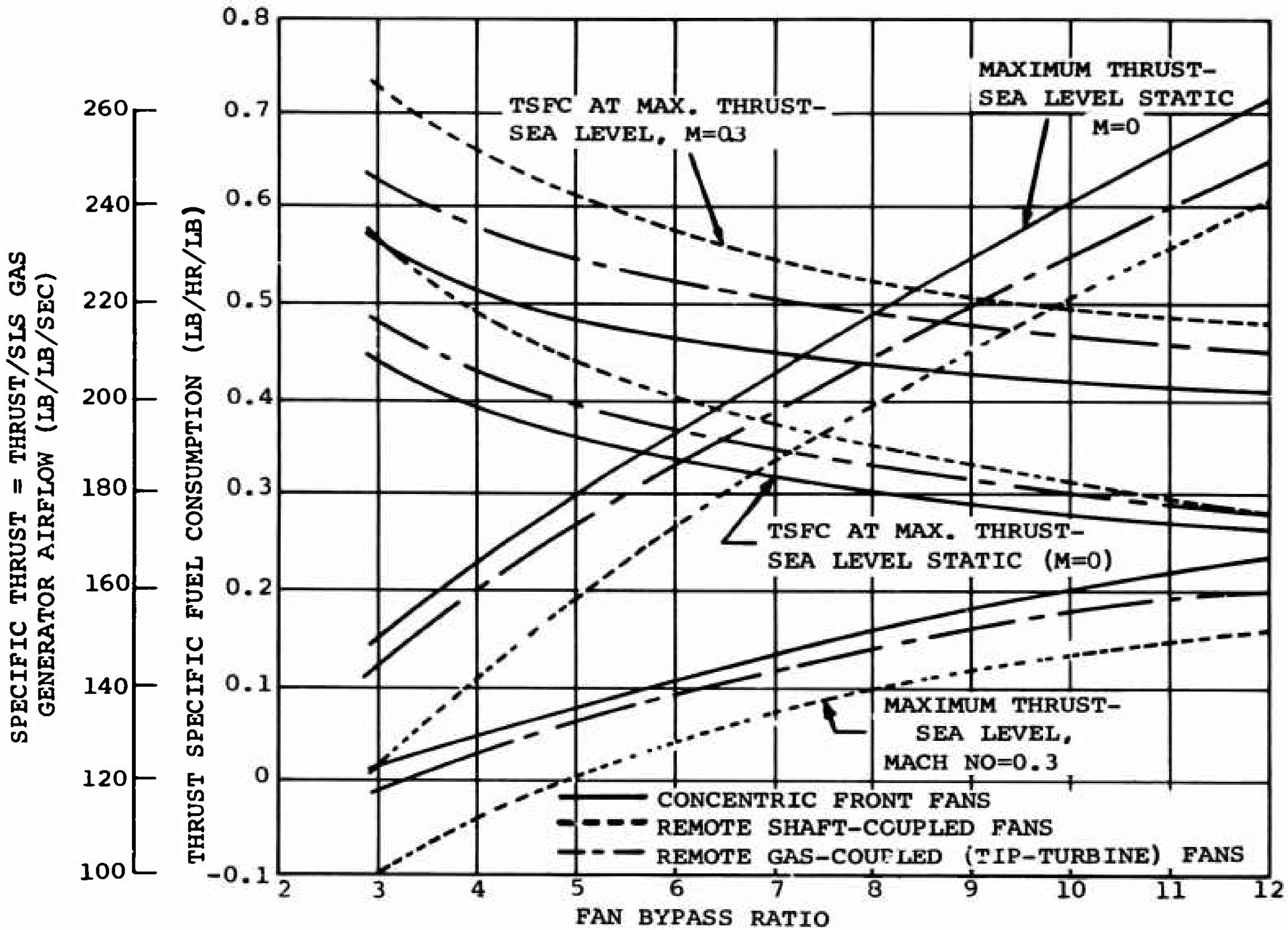


Figure 29. Comparison of the Performance (Uninstalled) of Various Propulsion Systems

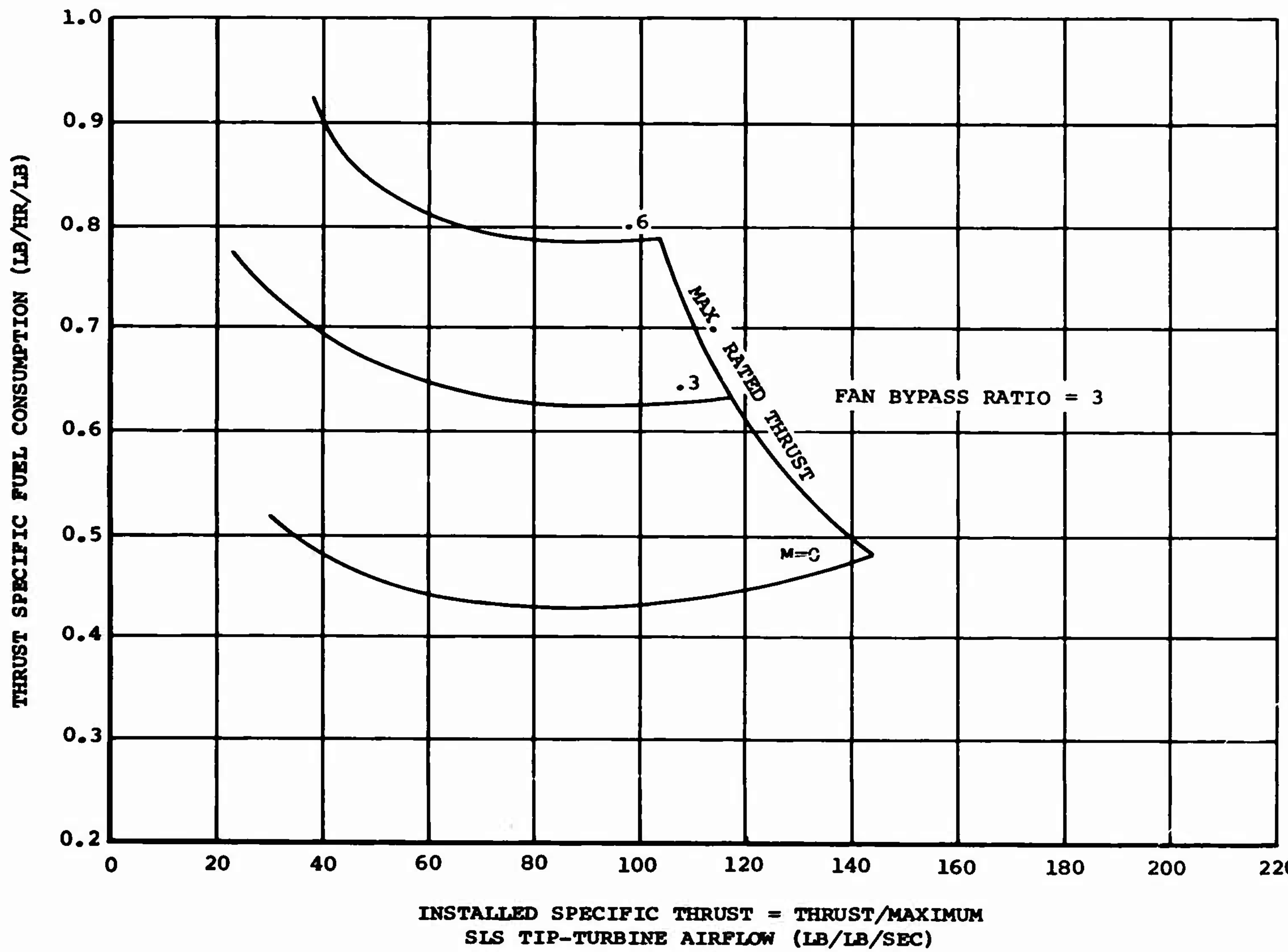


Figure 30. Installed Sea Level Standard Day Performance of Remote Gas-Coupled Tip-Turbine Fan System 1a (Sheet 1 of 4)

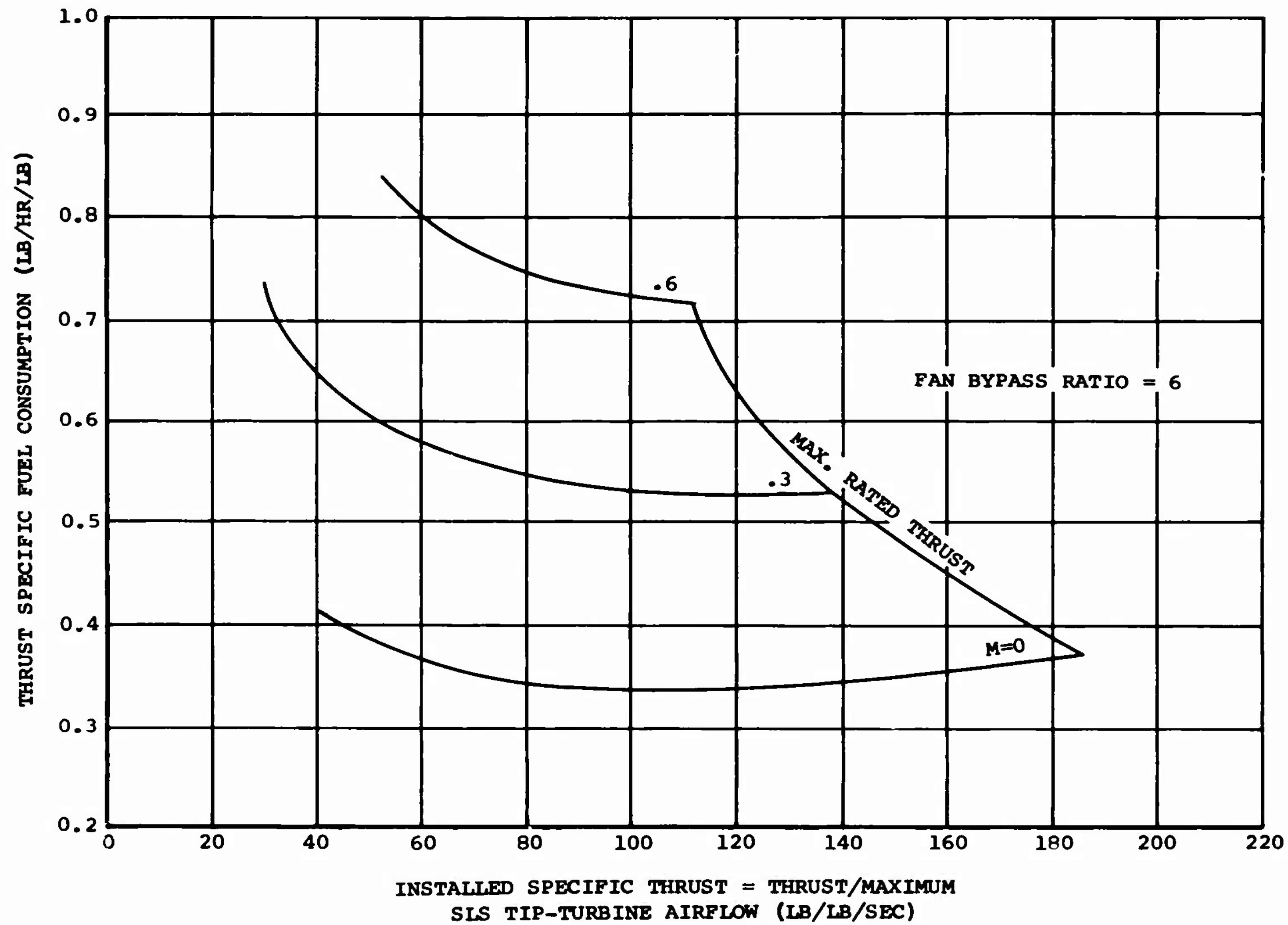


Figure 30. Installed Sea Level Standard Day Performance of Remote Gas-Coupled Tip-Turbine Fan System 1a (Sheet 2 of 4)

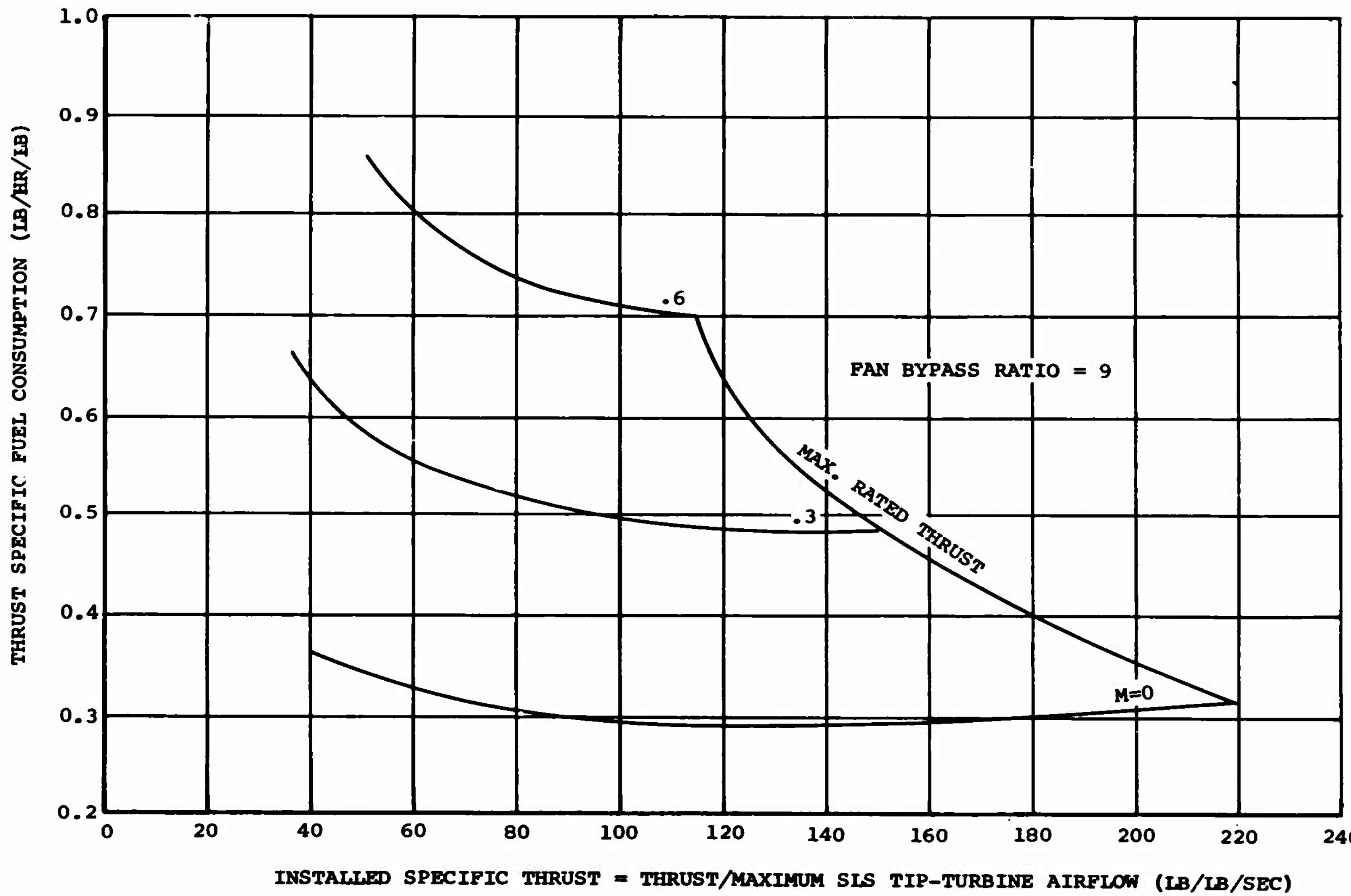


Figure 30. Installed Sea Level Standard Day Performance of Remote Gas-Coupled Tip-Turbine Fan System 1a (Sheet 3 of 4)

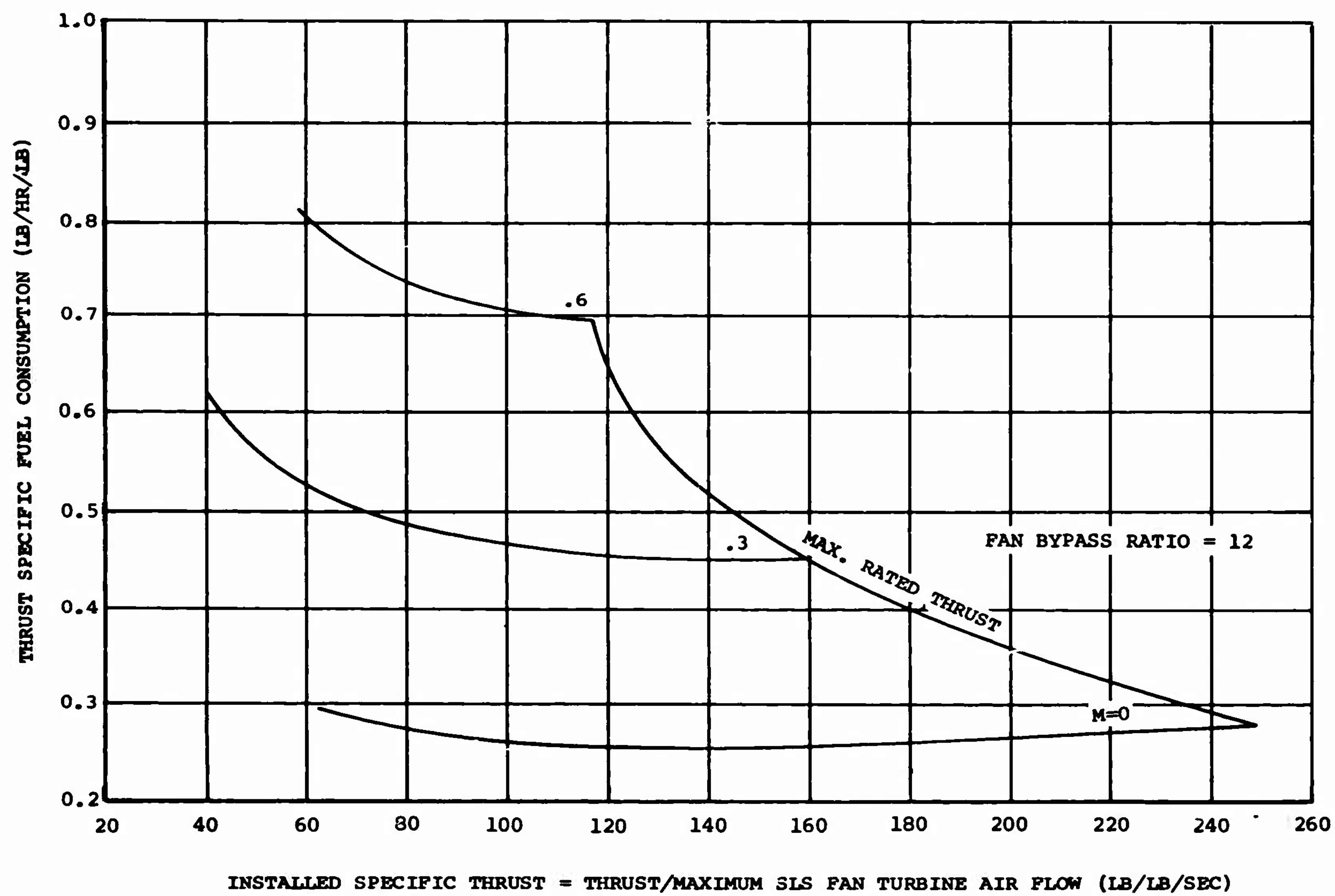


Figure 30. Installed Sea Level Standard Day Performance of Remote Gas-Coupled Tip-Turbine Fan System 1a (Sheet 4 of 4)

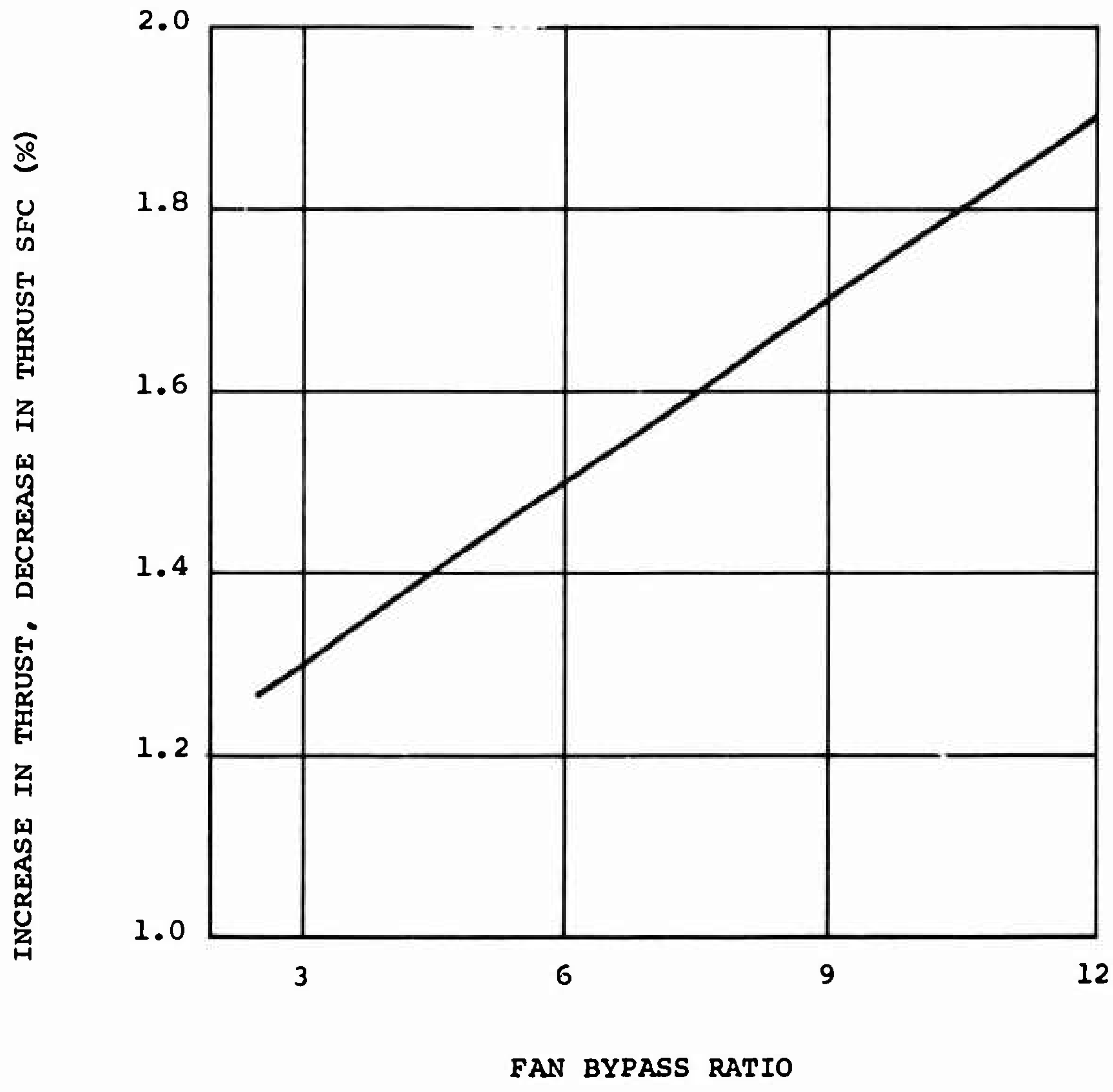


Figure 31. Factor to Correct Thrust and Thrust Specific Fuel Consumption of Tip-Turbine Fan-System (1a) to Axial-Turbine Fan-System Performance (1b)

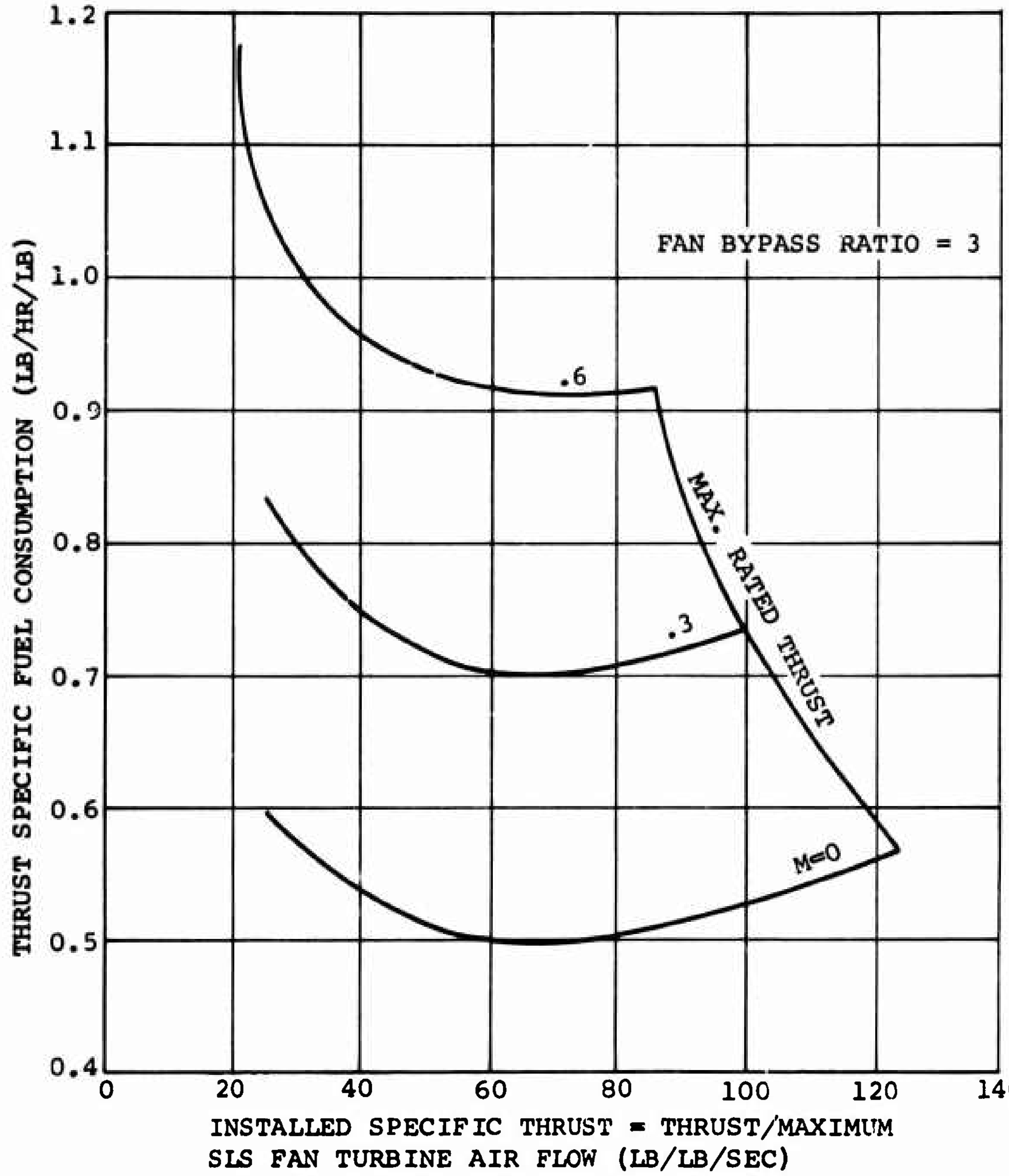


Figure 32. Installed Sea Level Standard Day Performance of Remote Shaft-Coupled Fan System 2a (Sheet 1 of 4)

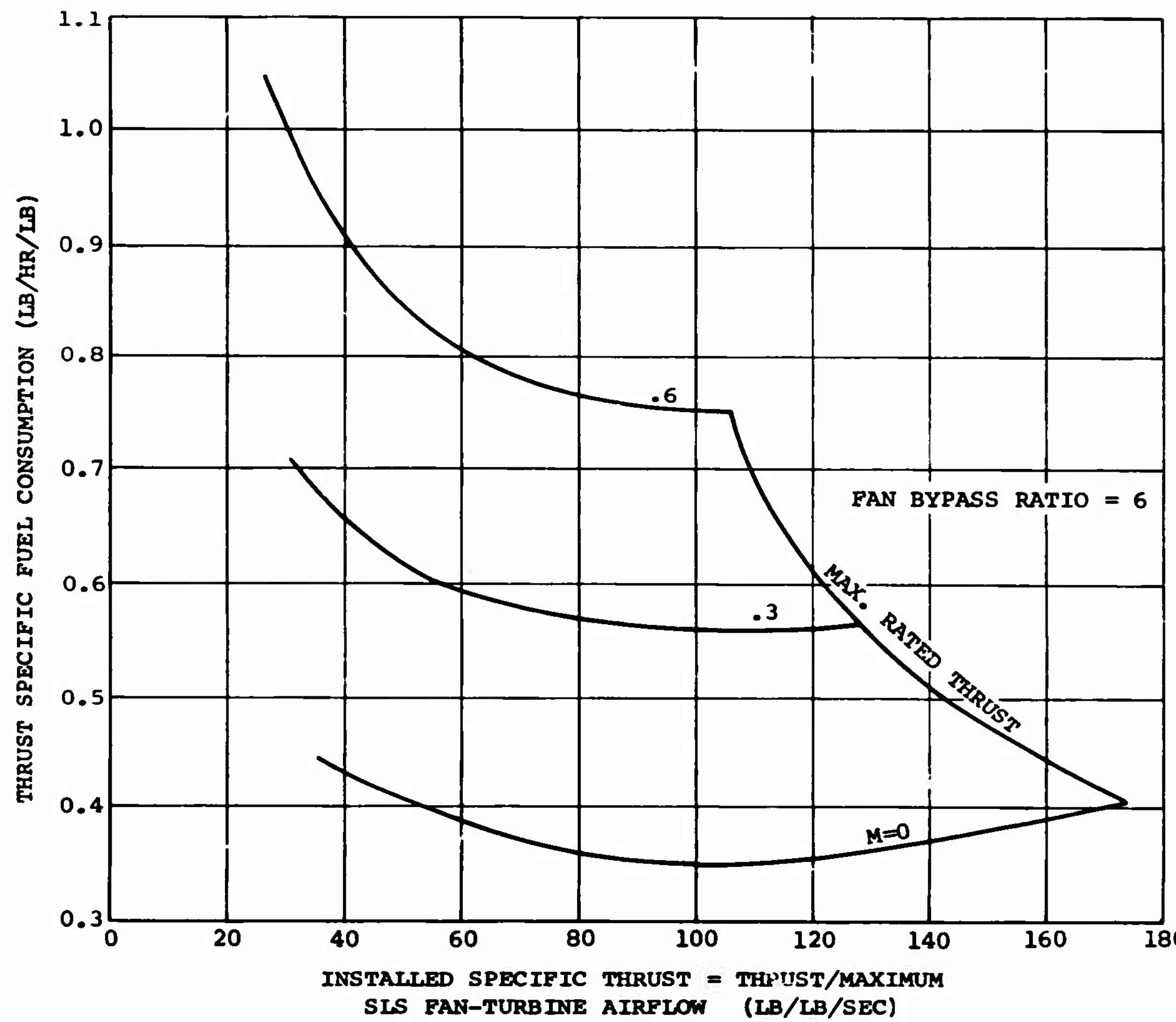


Figure 32. Installed Sea Level Standard Day Performance of Remote Shaft-Coupled Fan System 2a (Sheet 2 of 4)

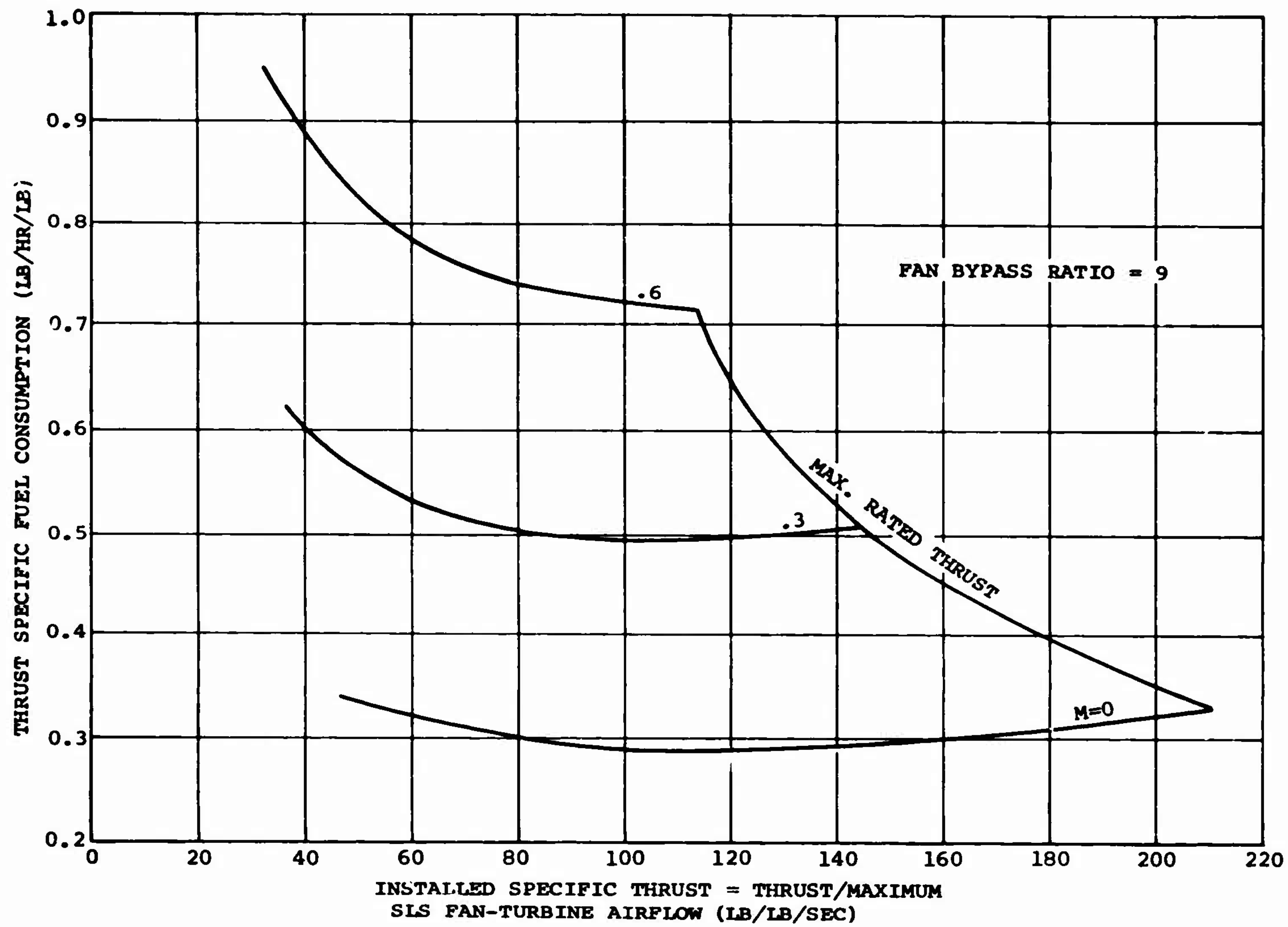


Figure 32. Installed Sea Level Standard Day Performance of Remote Shaft-Coupled Fan System 2a (Sheet 3 of 4)

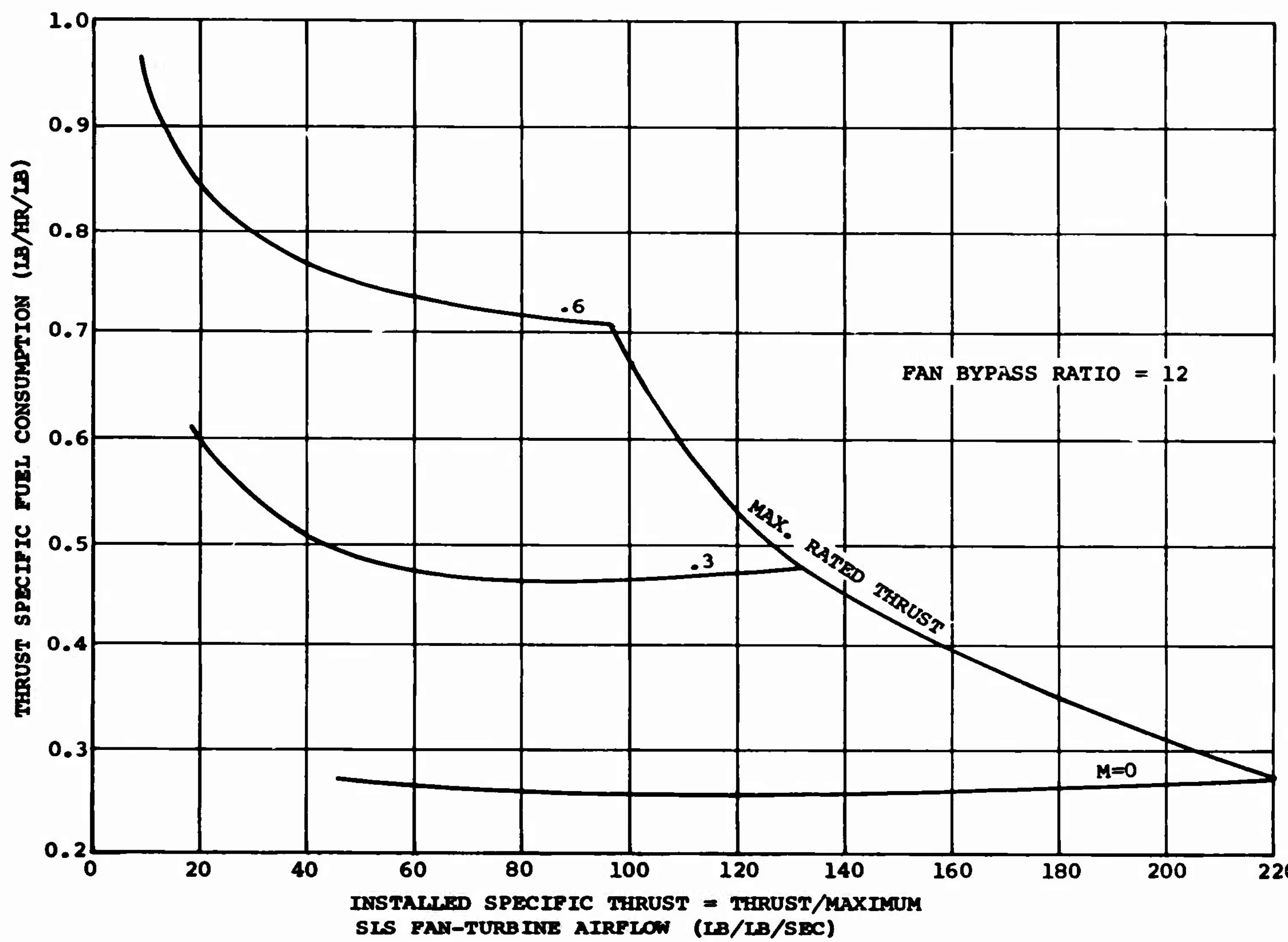


Figure 32. Installed Sea Level Standard Day Performance of Remote Shaft-Coupled Fan System 2a (Sheet 4 of 4)

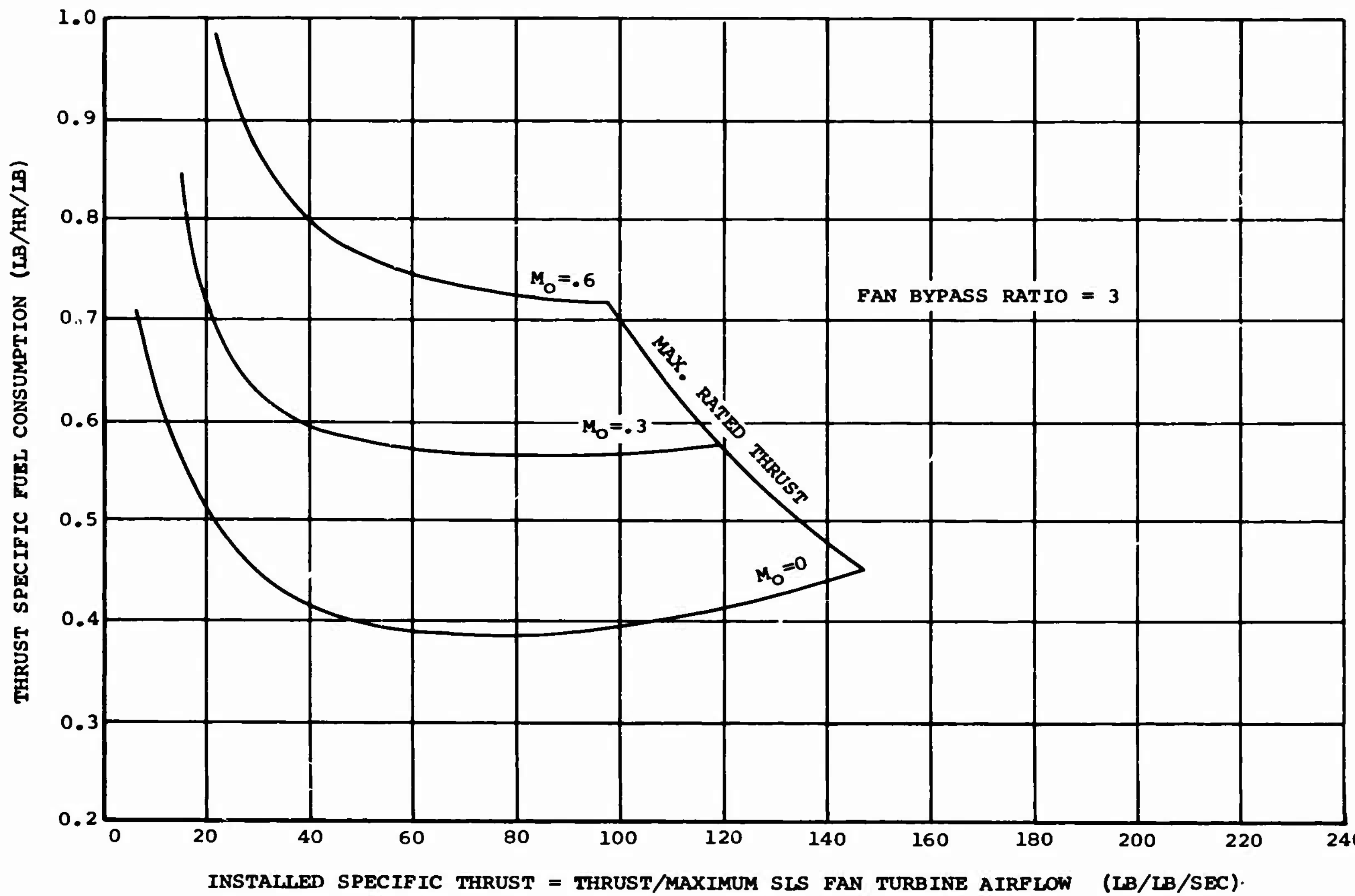


Figure 33. Installed Sea Level Standard Day Performance of Concentric Front Fan System 3 (Sheet 1 of 4)

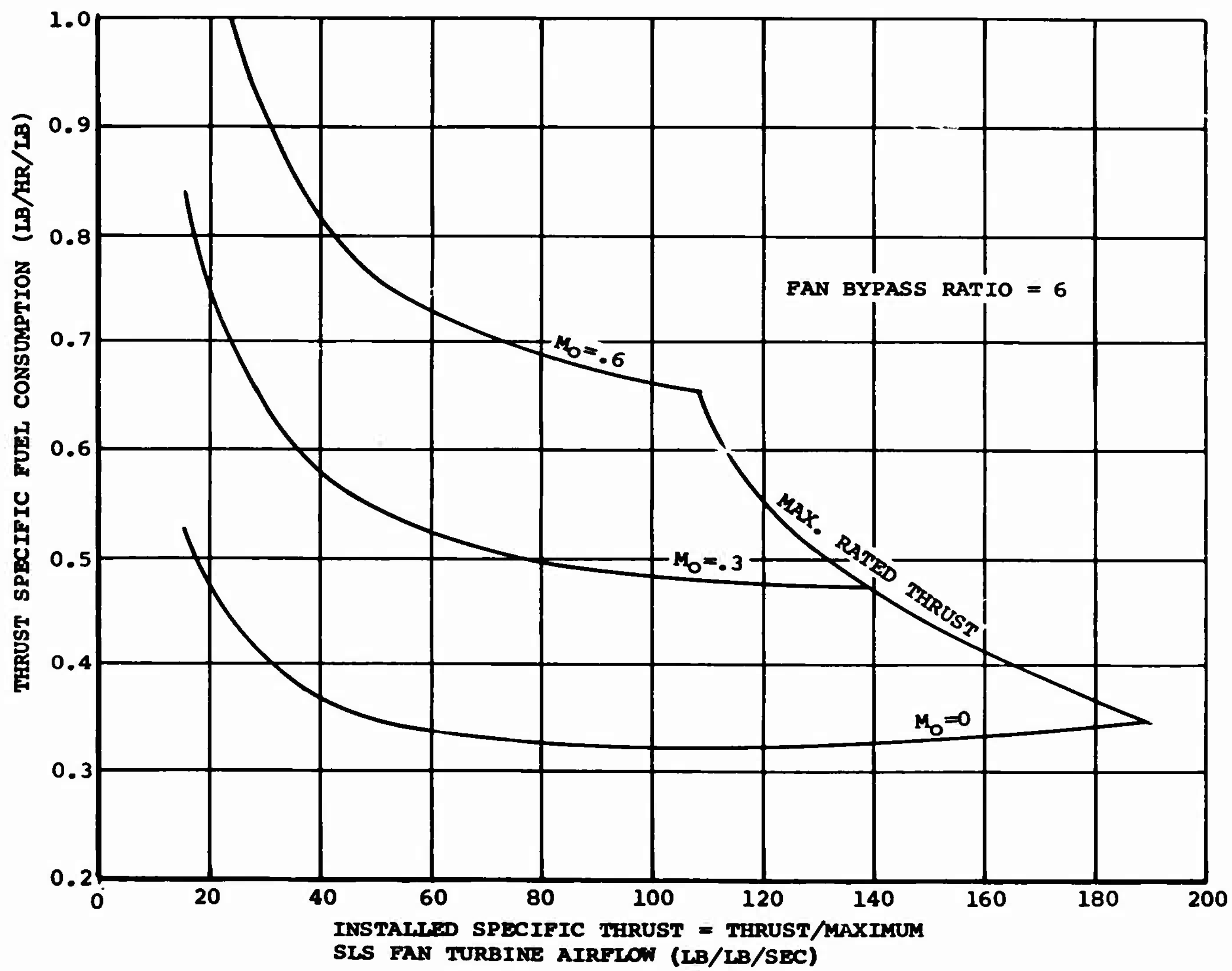


Figure 33. Installed Sea Level Standard Day Performance of Concentric Front Fan System 3 (Sheet 2 of 4)

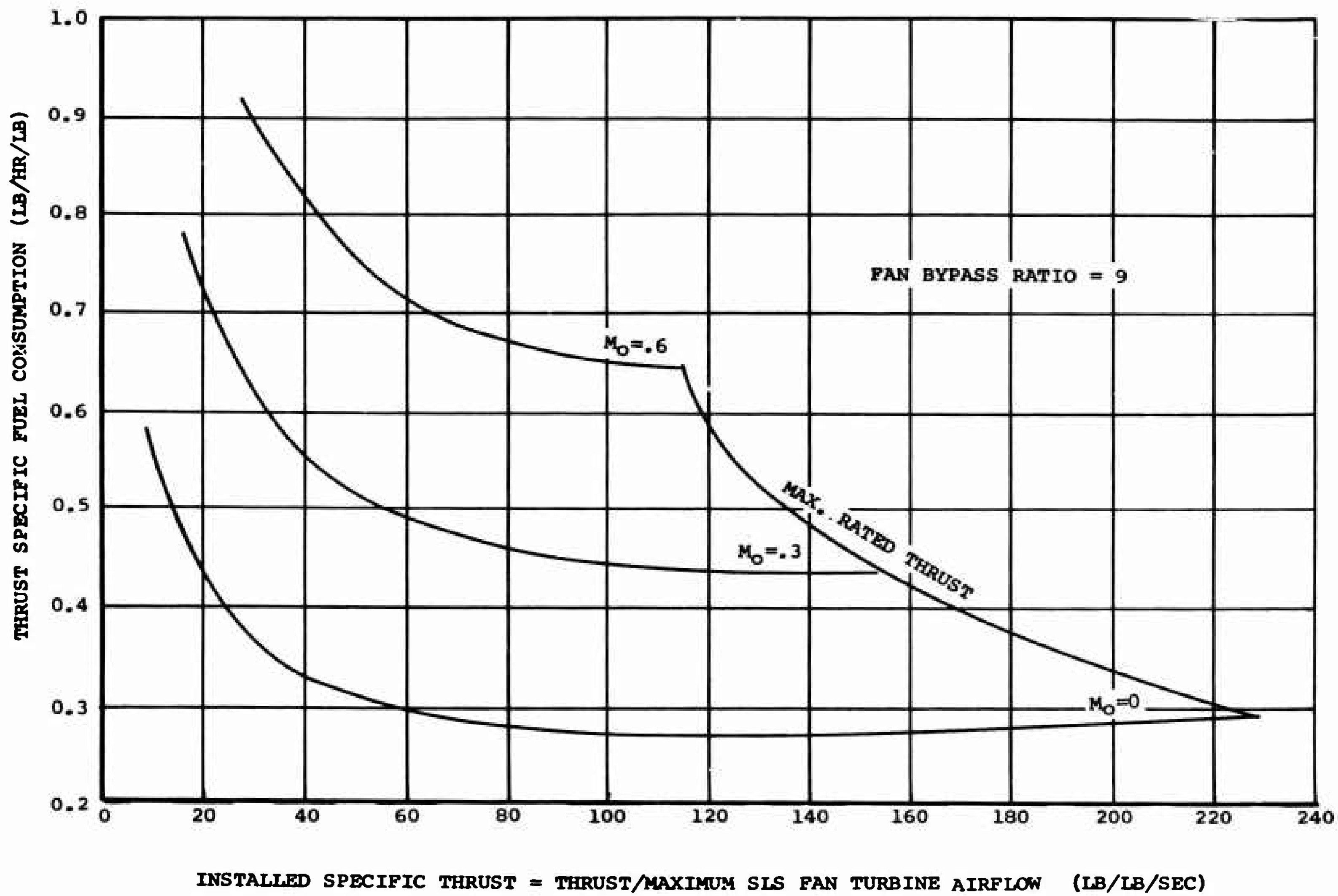


Figure 33. Installed Sea Level Standard Day Performance of Concentric Front Fan System 3 (Sheet 3 of 4)

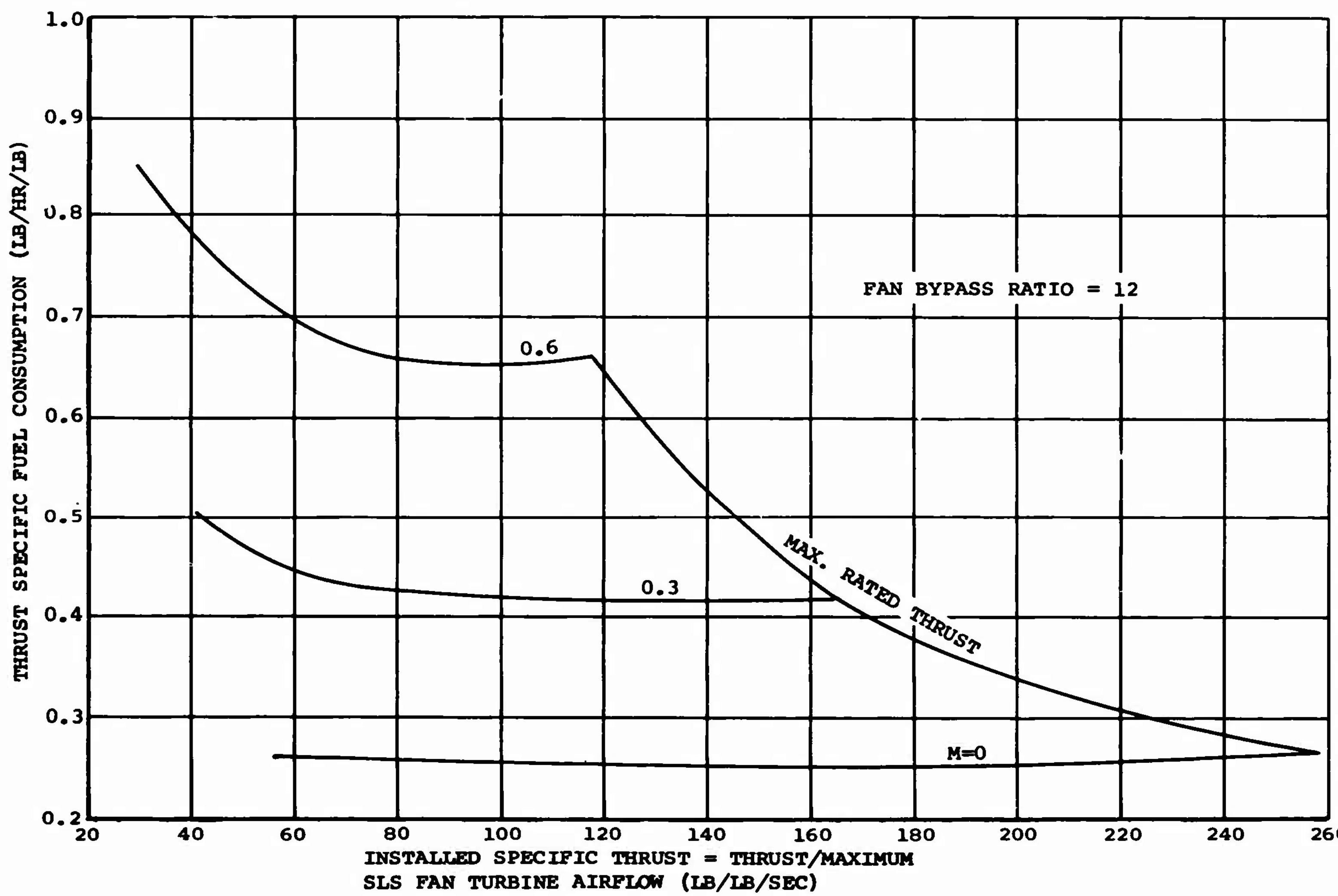


Figure 33. Installed Sea Level Standard Day Performance of Concentric Front Fan System 3 (Sheet 4 of 4)

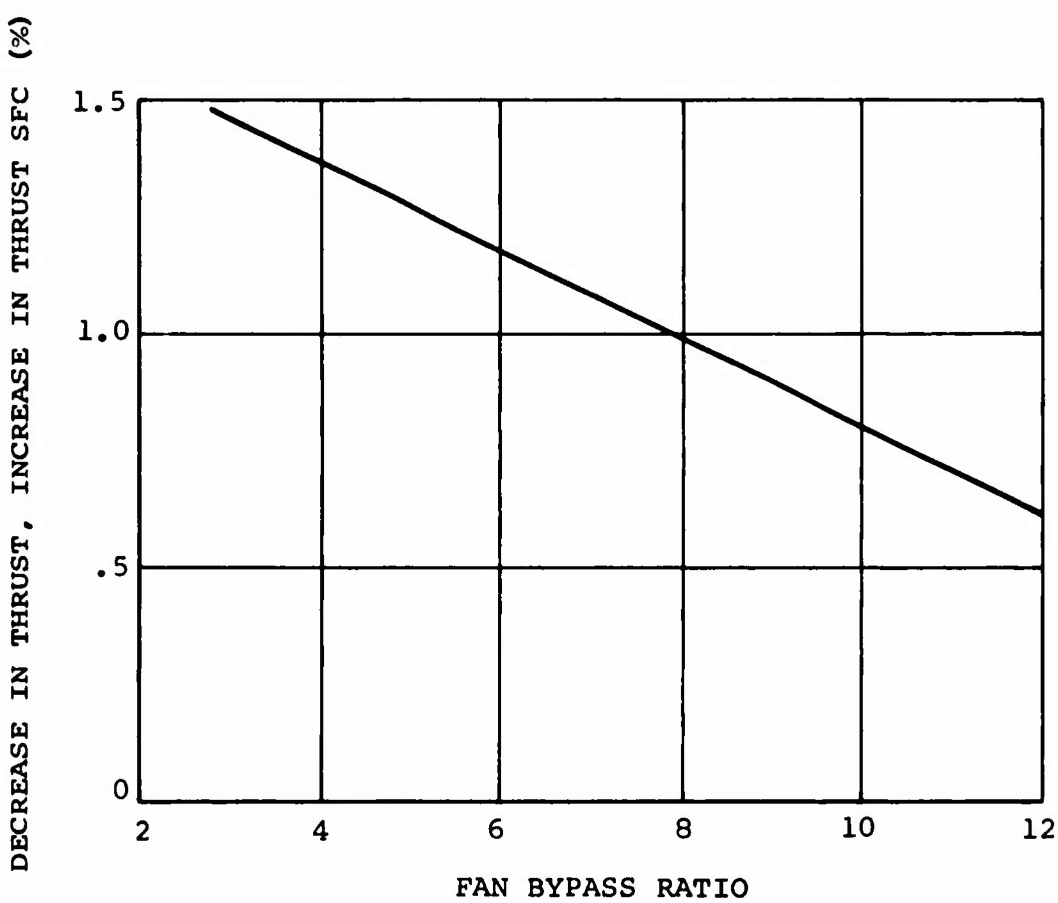


Figure 34. Factor to Correct Thrust and Thrust Specific Fuel Consumption of Concentric Front-Fan System (3) to Convertible Fan Performance (2b)

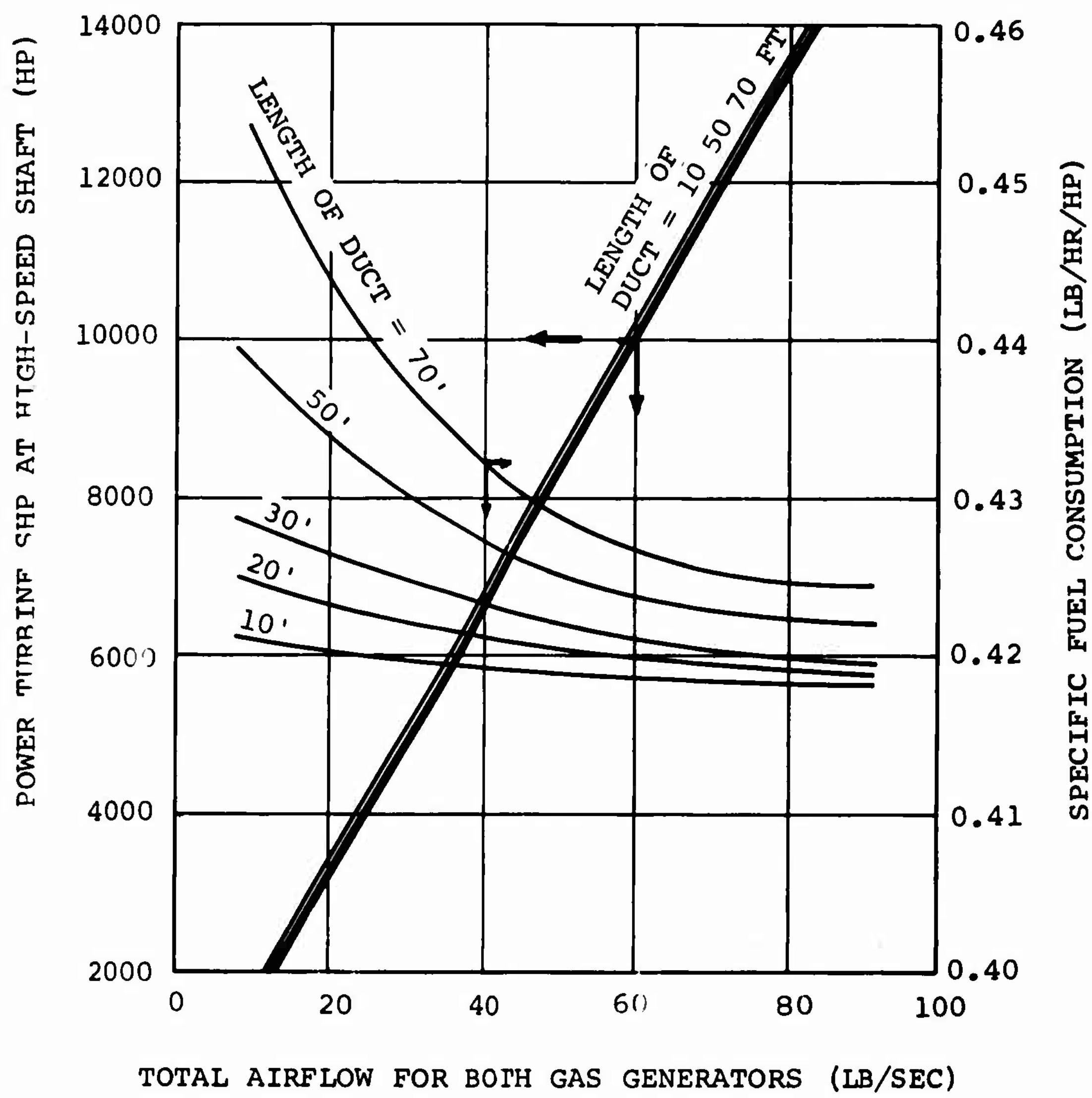


Figure 35. Remote Power Turbine Sea Level Static Maximum Horsepower and Specific Fuel Consumption (Systems 1a and 1b)

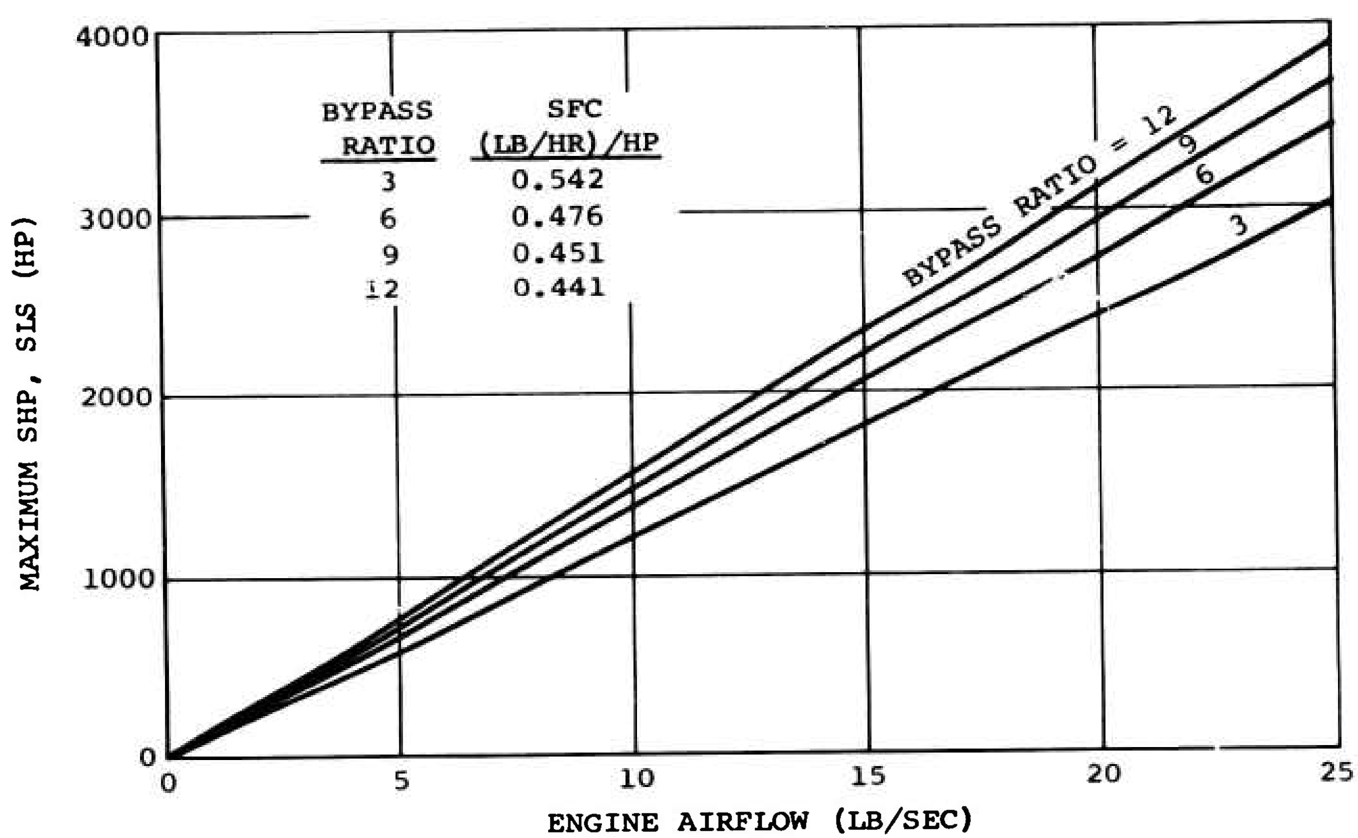


Figure 36. Convertible Engine Sea Level Static Maximum Horsepower and Specific Fuel Consumption

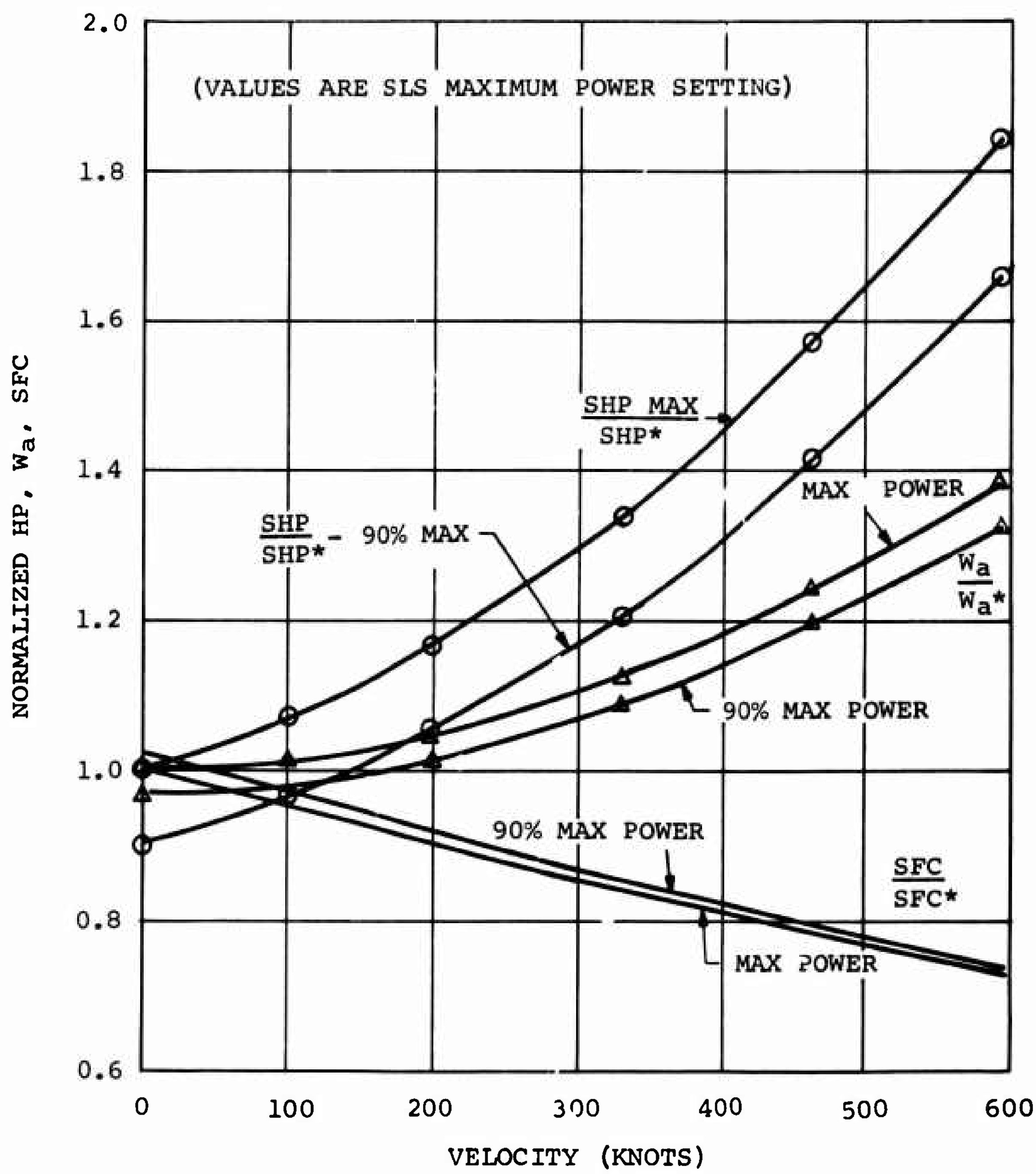


Figure 37. Shaft Engine Power, SFC, and Airflow at Maximum and 90% Maximum Power Settings

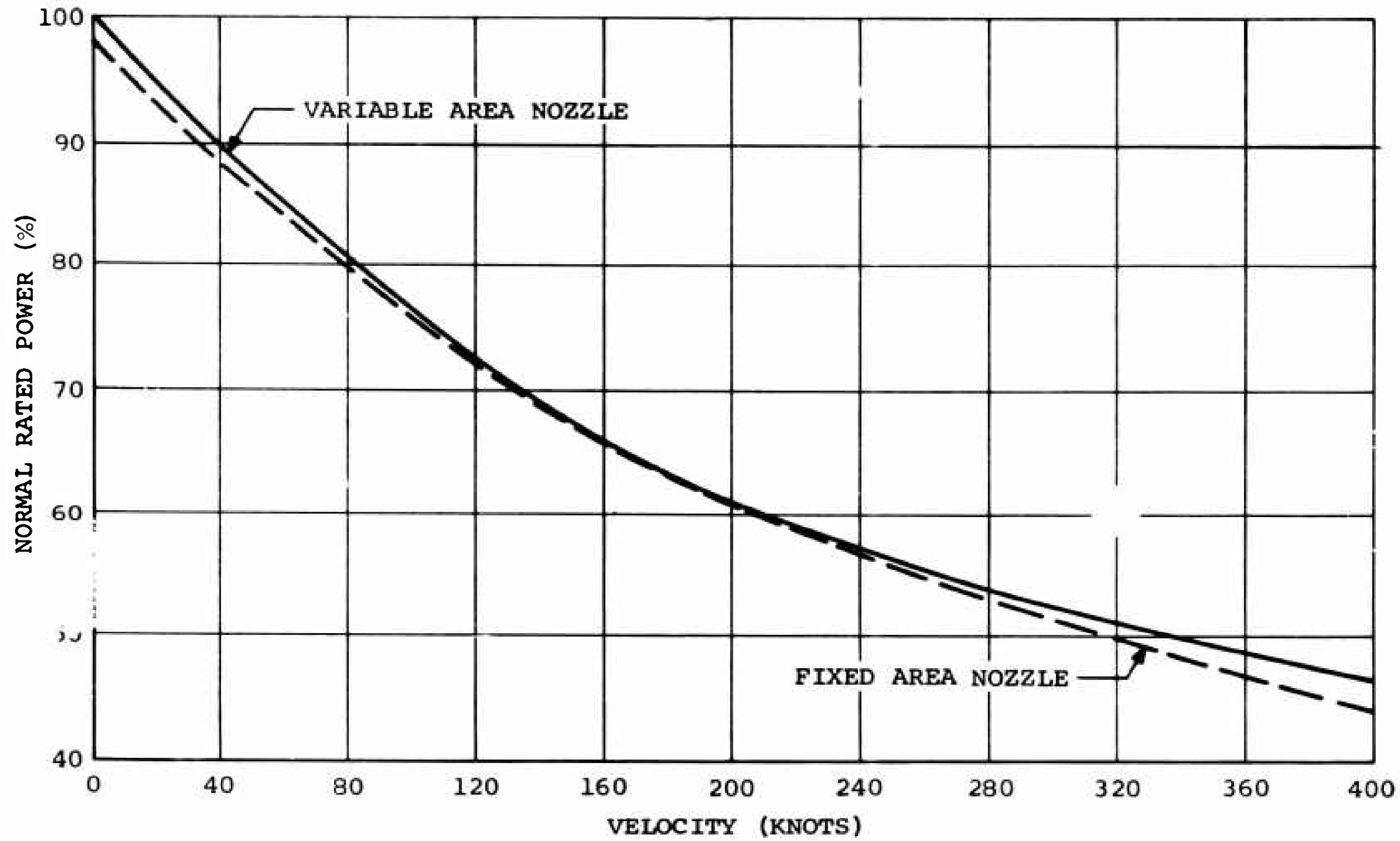


Figure 38. Effect of Fixed and Variab_e Area Exhaust Nozzle on Turboshaft Engir_e Performance

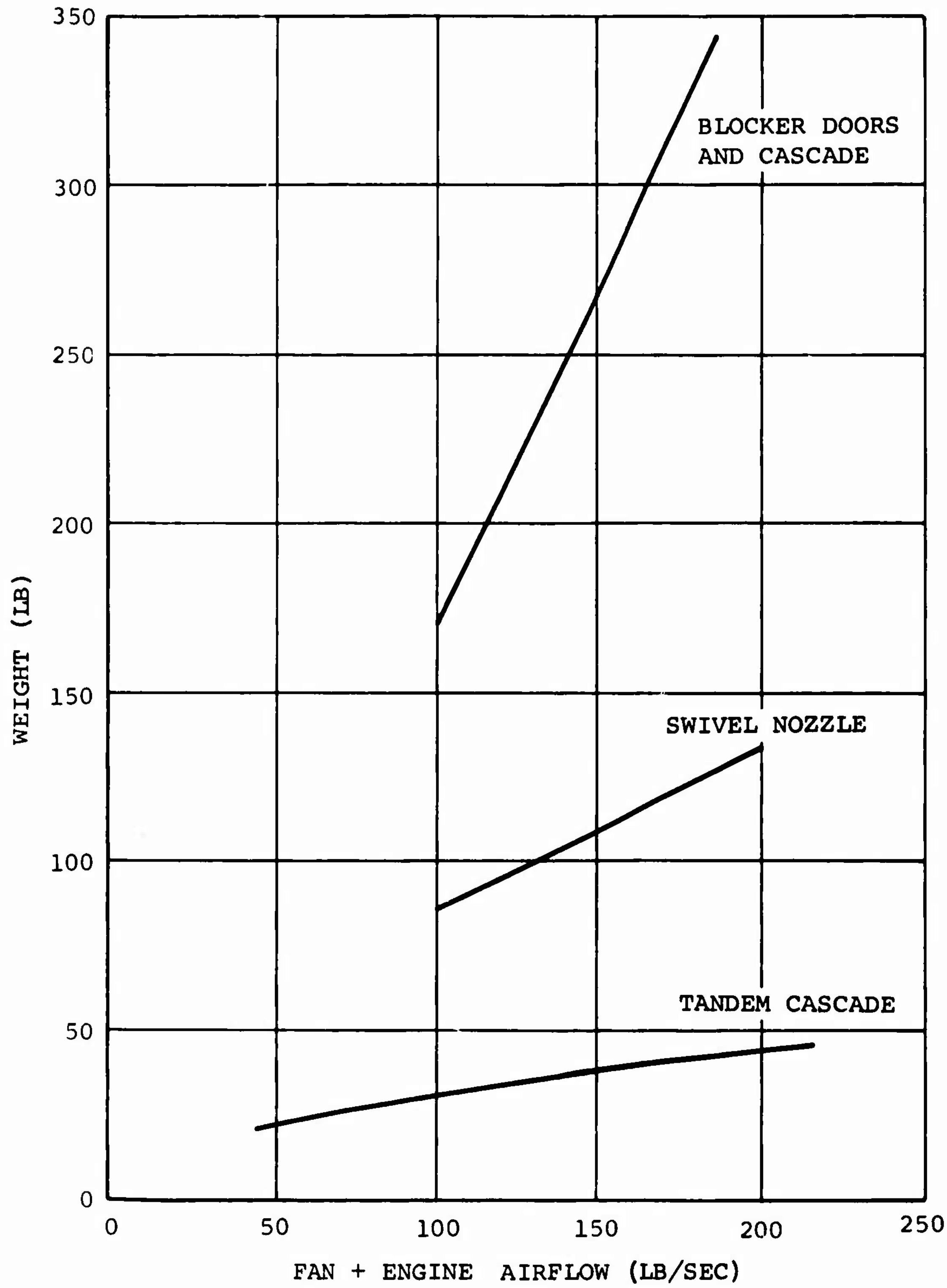


Figure 39. Weight of Thrust Vectoring Systems

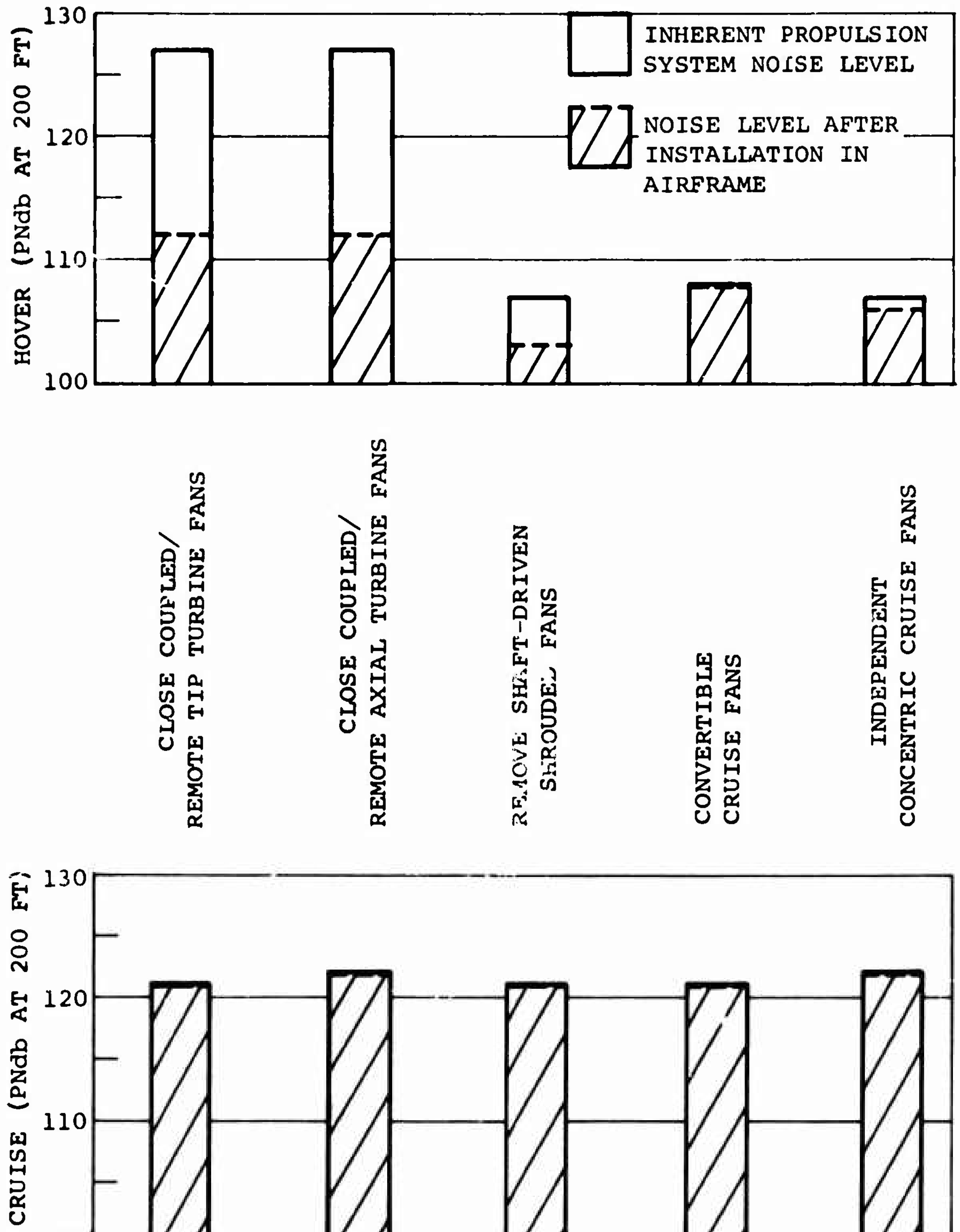
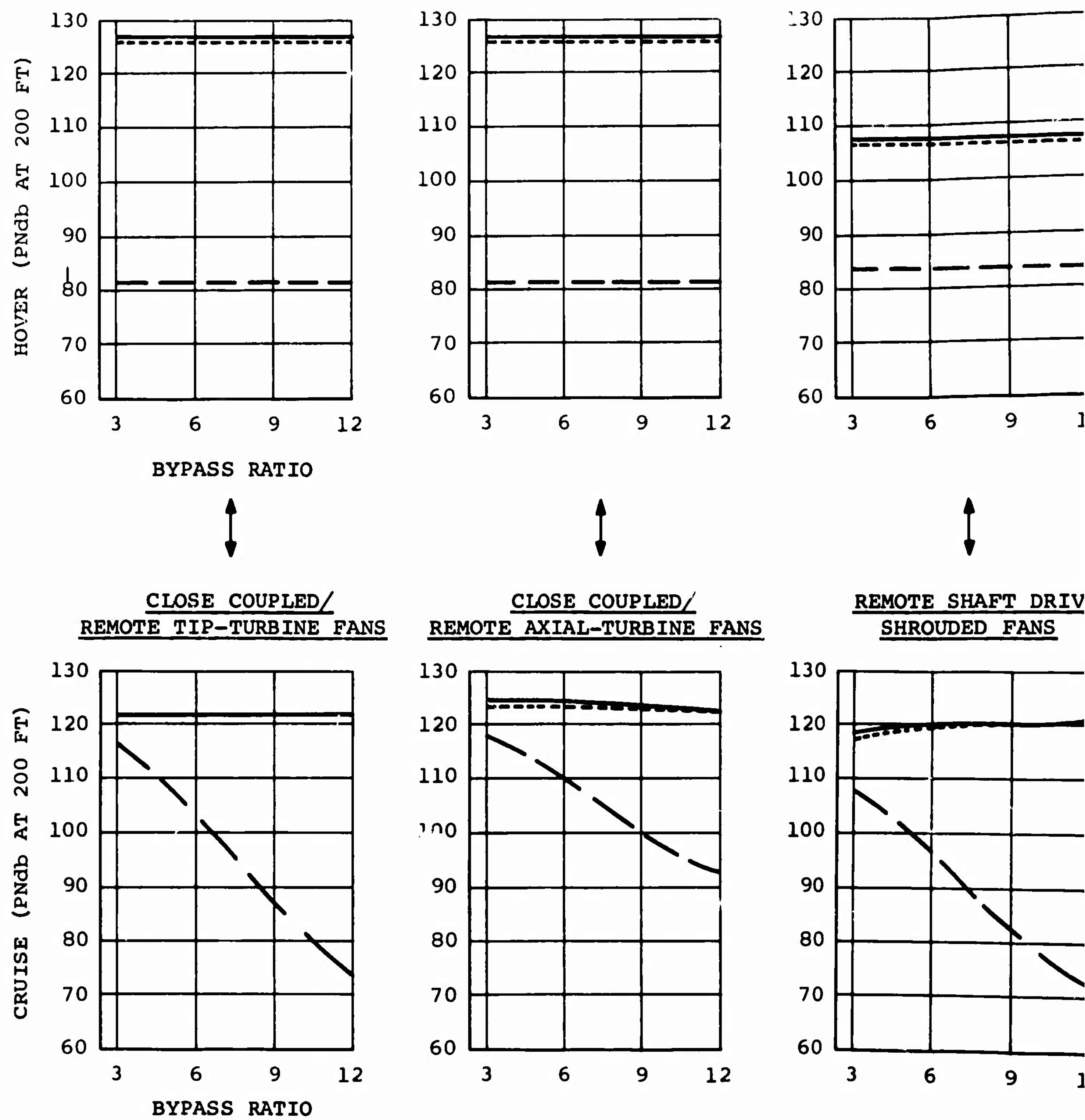


Figure 40. Optimum Cruise-Fan Propulsion System in Regard to Noise

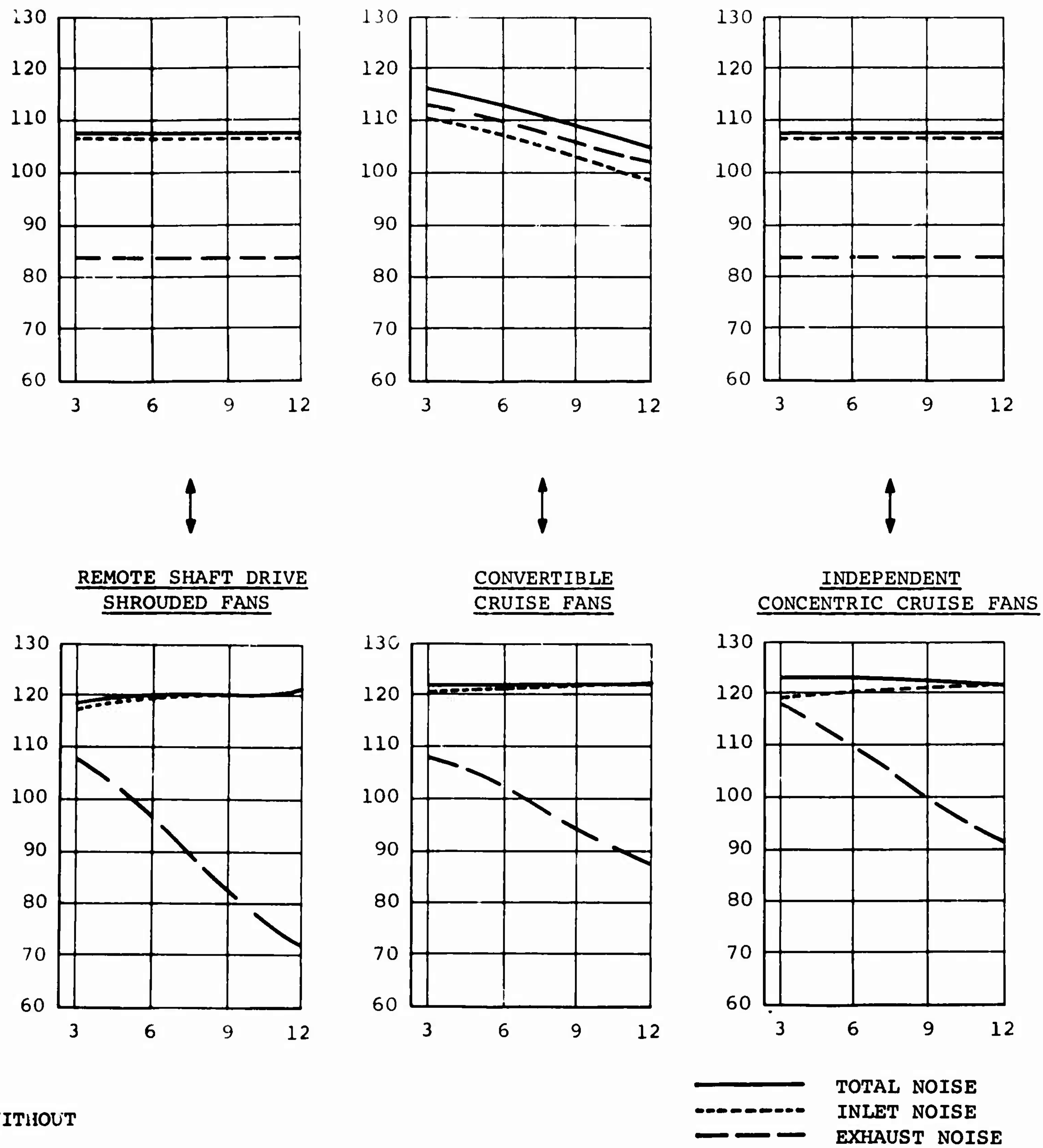


NOTE: NOISE LEVELS SHOWN ARE FOR PROPULSION SYSTEMS WITHOUT INSTALLATION EFFECTS

Figure 41. External Noise Characteristics

A

IS


REMOTE SHAFT DRIVE SHROUDED FANS
CONVERTIBLE CRUISE FANS
INDEPENDENT CONCENTRIC CRUISE FANS
TOTAL NOISE
INLET NOISE
EXHAUST NOISE

WITHOUT

B

COMPOUND/COMPOSITE CONFIGURATION STUDIES

Performance of this study required the generation of design data sufficient to define, on a preliminary parametric basis, the configuration and size characteristics of a broad spectrum of compound and composite aircraft. These vary in aircraft type, number of lifting rotors, and disc loading, as well as in the incorporation of five different lift/propulsion systems with various cruise fan bypass ratios. The variations are as follows:

Aircraft Types (3 variations): propulsion-unloaded compound, lift/propulsion-unloaded compound, and composite.

Number of Lifting Rotors (2 variations): single and tandem

Disc Loadings (3 variations): 5, 8, and 11 pounds per square foot.

Propulsion System Types (5 variations): Integrated gas-coupled fan systems in two variants (tip- and hub-driven), integrated shaft-coupled fan systems in two variants (remote and concentric), and an independent hover-cruise arrangement. The propulsion arrangements are designated respectively as systems 1a and 1b, 2a and 2b, and 3.

Fan Bypass Ratios (4 variations): 3, 6, 9, and 12.

Following is a discussion of the basic design and other study ground rules and the methods and background data used to define and scale the resulting 360 parametric aircraft of the study. Also included is a description of the various propulsion and drive systems. The data referenced herein, together with a series of point design layout investigations, is sufficient to provide a basis for predicting scaled aerodynamic drag and weight estimates for all aircraft in the study.

AIRCRAFT DESIGN STUDY GROUND RULES

The study ground rules of greatest importance for definition of the preliminary design configuration were:

1. Unpressurized, rear ramp-loaded internal cargo compartment size: 90 inches wide by 78 inches high by 30 feet long

with 78-inch head clearance for ramp entry at ground static conditions on all aircraft.

2. Twin engine installation (for the lift system) on all aircraft.

Other study ground rules affecting aerodynamic, structural, or weight characteristics include:

1. Constant design mission of 100-nautical-mile radius at sea level, and constant mission payload, out and return, of 6000 pounds.

2. Constant fixed-equipment weight of 2,935 pounds and fixed useful load of 750 pounds for all aircraft.

3. Constant flight-maneuver limit load factor of +3.0, -0.5.

4. Design gross weight and installed lift-system power defined by 6000-foot, 95°F hover capability carrying the useful load for the design mission.

Aircraft Design Configurations

Six basic aircraft configurations were employed in the study, all of which are advanced derivatives of current single and tandem rotor helicopters. Following is a short description of each aircraft type:

Tandem Rotor Propulsion-Unloaded Compound Aircraft - This type is a direct variant of operational tandem rotor helicopters. The rotors are thrust-unloaded by auxiliary propulsive fans to obtain higher speeds. Lifting and control forces are supplied by the rotors throughout the flight spectrum. The configuration selected is similar to current models except that cruise fans are used, the landing gear is retracted, contouring of the aircraft is revised, and advanced design changes are incorporated to reduce drag and weight.

Single Rotor Propulsion-Unloaded Compound Aircraft - This type is a derivative of the current single rotor helicopter type with tail antitorque rotor. Otherwise, the equivalent configuration description of the tandem rotor aircraft applies.

Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft - This aircraft type is another tandem rotor helicopter variant in which the rotors are not only thrust unloaded by propulsive fans but are lift unloaded in cruise flight by conventional wings to improve further the flight performance parameters. Rotors are slowed but not stopped in high-speed flight.

The selected tandem rotor aircraft configuration employs a conventional single wing and horizontal tail. The aft rotor pylon and vertical tail are one integral unit. Standard helicopter controls are used in hover and low-speed flight;with airplane-type control surfaces are utilized after conversion to cruising flight is made. Again, a drag cleanup is applied, and advanced state-of-the-art components are employed in the aircraft.

Single Rotor Lift/Propulsion-Unloaded Compound Aircraft - This vehicle is a single rotor helicopter version also incorporating a wing for unloading rotor lift as well as fans for rotor-thrust unloading in cruising flight. Tail surfaces are of the airplane type and operate similarly. Otherwise, the description noted above in the equivalent tandem rotor configuration applies.

Tandem Rotor Composite Aircraft - The tandem rotor composite aircraft configuration incorporates the greatest change from the pure tandem helicopter. It flies as a helicopter at speeds below transition. During the transition flight phase the rotors are stopped, folded, and stowed within fairings to reduce drag and to allow high cruise speeds in the pure airplane mode.

The selected tandem composite aircraft configuration incorporates conventional wing, tail, and surface controls, in addition to a retractable landing gear. The rear rotor pylon is employed as a vertical tail. A stowage bay for the folded forward rotor is provided along the top of the fuselage above the payload compartment. A fairing door system is provided to enclose the bay. In addition, a fairing assembly is provided for the stowed rear rotor which encloses the hub and the inboard portions of the folded blades. Outboard sections of these blades remain together in horizontal trail.

The aft fairing assembly includes a base section fixed to the top of the rear pylon which provides clearance for rotor operation in all conditions down to maximum blade droop angle.

It also includes an airfoil-shaped fairing of low aspect ratio sized to enclose the stowed rotor in high-speed flight. The fairing can be retracted to a lowered position during the regime of rotor operation. The top section of this fairing consists of a segmented panel of the roll-top-desk type, designed to slide into the nose section of the fairing prior to retraction so as to avoid interference with the rear rotor during the retraction process.

Single Rotor Composite Aircraft - The single rotor composite type employed for this study is a further adaptation of the conventional single rotor helicopter employing cruise fan propulsion, conventional wings and tail surfaces, and a retractable landing gear for high-speed flight. In this regime the main rotor is stopped, folded, and partially faired over, and the tail rotor is stopped. A review of the possibility of also folding and stowing the tail rotor was conducted; however, any gains that might be achieved appeared uncertain; therefore, a trade-off analysis was considered to be beyond the scope of this study.

The fairing system for the main rotor of this type is identical in concept to that employed on the rear rotor of the tandem composite aircraft.

AIRFRAME PARAMETRIC DESIGN DATA

Main (Lifting) Rotors - The range of required rotor diameters for this study was equal to 50 to 100 feet for single rotor aircraft and 35 to 80 feet for tandem rotor aircraft. The disc loading used herein is based on gross weight rather than on gross thrust. The range of design gross weights estimated to encompass all 360 aircraft design points was set at 20,000 to 50,000 pounds based on early approximations of potential best and worst combinations of aircraft lift-drag ratios, propulsive efficiencies, specific fuel consumptions, and useful load efficiencies.

The number of blades per rotor assumed for single and tandem aircraft of all types is shown plotted against rotor diameter and disc loading in Figure 42. Based on examination of the required study diameter range, together with review of practical blade chords and resulting aspect ratios at the solidities meeting the 6000-foot, 95°F hover requirement, three-bladed rotors were selected for all tandem rotor aircraft. Three, four, or five rotor blades, depending on diameter, were

selected on a similar basis for single rotor configurations. For the single rotor composite aircraft type the number of blades was limited to three to effect the required blade folding and stowage for high-speed cruise flight.

The required blade chord for all main rotors during hover flight was based on the 6000-foot, 95°F hover condition with all rotors operating at a tip speed of 750 feet per second and $\bar{c}_\ell = 0.66$, and was derived as follows:

$$c = \frac{2\sigma_R S}{b_T D} \quad \text{and} \quad \sigma_R = \frac{6(GW/S)(1 + DL)}{(\sigma\rho_o)\,\bar{c}_\ell V_T^2}$$

$$\text{or} \quad c = K(GW/b_T D)(1 + DL)$$

where

c = required blade chord (ft)

σ_R = rotor solidity

S = total disc area (ft^2)

b_T = total blades/aircraft

D = main rotor diameter (ft)

GW = gross weight (lb)

$$k = \frac{12}{(\sigma\rho_o)\,\bar{c}_\ell V_T^2} = \text{constant} = 0.018$$

DL = download (percent thrust)

σ = density ratio

ρ_o = sea level standard density ($slugs/ft^3$)

$\bar{c}_\ell$ = average lift coefficient

V_T = rotor tip speed (ft/sec)

These rotor chords were employed for scaling of all lift/propulsion-unloaded and composite aircraft in the study. For propulsion-unloaded aircraft, rotor chords were based on the requirements of the design cruise flight condition.

The thickness ratio for all rotor blades is assumed to be 11 percent. Blade folding for ground stowage purposes alone was not assumed as a requirement for any of the aircraft.

The lifting and propulsive capability of rotors is limited by considerations other than power. Therefore, to avoid undesirable rotor phenomena associated with blade stall such as stall flutter, flapping divergence, or flap lag instability, a limiting criterion for rotor operation was established for all the study aircraft. Use of the criterion ensures that rotor lift does not exceed values determined as limits from the Boeing analysis based on correlation of flight and wind tunnel test data. This criterion is in the form of a limiting rotor vertical force coefficient variation with an advance ratio incorporating appropriate deviations for changes in propulsive force requirements.

A further constraint upon the rotor propulsive capability of the study aircraft was the imposition of a maximum rotor control axis tilt angle (plane of no feathering angle) of -15°. This angle, taken as a limiting case of the sum of aircraft nose-down flight attitude angle, permissible effective rotor shaft tilt angle, and peak longitudinal cyclic pitch control angle, was established on the basis of practical aircraft design considerations.

Wings - The wing loading for lift/propulsion-unloaded compound aircraft was selected on the basis of achieving a desirable operating lift coefficient at the design sea level cruise speed with rotor system unloaded, and to provide margin for maneuver capability. The wing loading for composite types was selected based on an end-of-conversion speed of 125 knots at sea level with a 20-percent margin allowed on stall to provide for potential gust conditions. Single-slotted flaps and drooped ailerons are estimated to provide an available maximum lift coefficient of well over 2.0 in this case. In all cases, the wing area specified includes the fuselage-blanketed area, but does not include the area of any inboard stubs, gloves, or other provisions for landing gear.

All wings were designed for an aspect ratio (AR) of 6 and a constant thickness ratio (t/c) of 15 percent along the span. The wing leading edge sweep angle is 20 degrees, except in tandem rotor aircraft where 10 to 15 degrees of sweep angle was employed to minimize interference problems with other airframe components and with the ground when tilted. The taper ratio

is 0.5 to 1 except where design layout dictated a reduction to 0.35 to 1, again for clearance reasons.

Major considerations as to vertical location of the wing on the fuselage in lift/propulsion-unloaded compound and composite aircraft were the necessity for minimum download on the wing from rotor wash in hover flight, adaptability of the aircraft for water landing capability as a normal operational procedure, and avoidance of potential integration problems of propulsion components.

A high wing was considered to be necessary for water landing capability to eliminate the danger of wing structural damage from water impact. It was felt that safe water operation could be effected with a high tilting wing as well as with a high nontilting wing, assuming provisions are made for adequate lateral stability in the water. A low wing position appeared to be improper for water landings.

A tilting wing minimized download in hover flight, particularly in tandem rotor cases with large disc overlap. However, to tilt the wing and to provide adequate ground and rotor clearances, the wing was constrained to a high position on the fuselage.

Possible forward location of propulsion elements mitigated against selection of a high wing location due to the possibility of hot gas impingement directly on the wing surfaces. Possible aft location of propulsion systems appeared commensurate with a high wing location, assuming that cruise fans could be located to avoid undesirable wing wake effects on inlet performance.

In winged tandem rotor configurations, therefore, the selection of a high wing location was made. Also, the wing tilts except in scaling cases where little or no download improvement is obtained because of rotor disc separation. Modification for operational water capability is therefore possible in these vehicles.

In winged single rotor aircraft, selection of a fixed low wing was made, since balance and foreign object damage criteria required a forward and high location of cruise fans. Therefore, tilting was not possible or desirable from a download standpoint. However, the capability for normal operational water landings is lost in these cases.

Tail Surfaces and Tail Rotors - The scaling data for sizing tail rotors of single rotor aircraft is shown in Figures 43 and 44. Figure 43 gives an estimate of minimum tail rotor diameter to absorb not more than 10 percent of the total power available to main and tail rotors under conditions of hover at 6000 feet, 95°F. This information was gained from in-house preliminary design reference data for the selected study conditions of 750 feet per second tip speed of both main and tail rotor. Figure 44 was based on an interpolated value of rotor solidity from the same data, and gives tail rotor blade chord as a function of rotor diameter and number of blades. In the case of the single rotor composite aircraft configuration, the decision was made to stop and lock the tail rotor, and the number of blades was restricted to three. The stopped position for the three-bladed tail rotor of composite aircraft was selected so that one blade was oriented directly forward and the other two aft. This position appears to be best since the dynamic airloads and aeroelastic deflection will be minimized.

Scaling data for tail surface areas and other basic tail geometry selected for all aircraft types are given in Figure 45. The area of the single horizontal stabilizing surface employed on the side opposite to the tail rotor in single rotor propulsion unloaded aircraft is given as a function of tail rotor diameter, and is based on a review of typical single rotor aircraft. The horizontal tail area of the single rotor lift/propulsion-unloaded and composite aircraft types is shown as a direct function of wing area and an inverse function of tail arm, where the arm varies directly with the main rotor diameter. The constants of proportionality were gained from previous studies of similar aircraft types and are expressed as follows:

$$S_{T_H} = 8.57(S_w/\ell_t) = 8.57(S_w/0.565D) = 15.2\ (S_w/D)$$

where

S_{T_H} = horizontal tail area (ft^2)

ℓt = tail arm, wing C.P. to tail C.P. (ft)

S_w = wing area (ft^2)

D = diameter of main rotor (ft)

The horizontal tail area scaling for the tandem rotor lift/propulsion-unloaded and composite types of aircraft is also given in Figure 45, again as a function of wing area and tail

arm. Since the longitudinal spacing of rotor centers is fixed in the basic case, with cargo compartment length constant, the tail arm below rotor diameters of 63 feet is a fixed value and the tail area is a function only of wing area. In cases where rotor diameters exceed this value, a constant 34-percent value of rotor blade overlap has been assumed, and tail arm increases proportionately, with a corresponding decrease in horizontal tail area requirement. Though in some cases of gas-coupled propulsion-system installation scaling involving large airflows, the fuselage length increases slightly from this basic case in proportion to propulsion components length, a corresponding increase in tail arm does not occur due to a requirement for a slight rearward movement of the wing for aircraft balance purposes. Design checks have indicated that the tail arm employed for scaling in the basic case is still sufficiently accurate and no change has been made to account for the effect of high-airflow propulsion installations. The expression for tail area employed is:

$$S_{T_H} = 8.57 \frac{S_W}{26.7 + (D - 63)}$$

where
S_W = wing area (ft^2)

D = main rotor diameter (ft)

In all cases, the tail area specified includes the portion blanketed by the fuselage. Horizontal tail surfaces are not tilted in any of the aircraft.

Vertical tail surface scaling assumptions for single rotor aircraft types are also indicated in Figure 45. The proportions of these surfaces have been taken from a survey of actual aircraft and from previous design studies. The size of the rear pylon and vertical tail surface of tandem rotor aircraft was found to be a function of many overall airframe and propulsion system integration factors; therefore, no simple scaling expression could be set forth. As a result, weight and aerodynamics estimates were, in this case, made up on the basis of design layout investigations.

<u>Fuselage</u> - Basic fuselage sections of all aircraft were determined by the required payload compartment height and width, and modified as required to accommodate landing gear, wing, and propulsion system components. Fuselage length scaling is shown in Figure 46. For all single rotor aircraft, the minimum

fuselage length required is a function of the sum of main and tail rotor diameters. With constant payload compartment and nose section length, the expression for single rotor aircraft overall length is:

$$L_F = 26 + 0.5(D + D_T)$$

where

L_F = fuselage length (ft)

D = main rotor diameter (ft)

D_T = tail rotor diameter (ft)

The contour of the fuselage in side elevation is governed by the head clearance requirement aft of the cargo ramp and by clearance with rotor blades in the maximum drooped position. Contouring of the fuselage in plan is controlled by the requirement of full width to the aft end of the cargo ramp. For single rotor aircraft with values of $D + D_T$ equal to or less than approximately 84 feet, the fuselage ends in a horizontal knife-edge at the tail with constant width due to insufficient length for tapering in plan aft of the cargo ramp. For values of $D + D_T$ greater than 84 feet, a taper in planform can be effected.

The basic length of fuselages of tandem rotor aircraft is a constant value except where modified by factors of maximum allowable rotor blade overlap and length of propulsion system components as determined by design airflows in the gas-coupled arrangements. The scaling in Figure 46 takes these factors into account. The minimum fuselage length is 58 feet for all aircraft except those with gas-coupled propulsion systems up to the rotor diameter where a constant 34-percent overlap factor is assumed. Fuselage length then increases proportionately. Figure 47 indicates the variation of overlap against rotor diameter used in the study. In the case of gas-coupled propulsion systems 1a and 1b, use of the most desirable location for propulsion components dictated a moderate fuselage length increase over basic values for some scaling cases. Propulsion system length then varies as a function of design airflows, and the fuselage length (Figure 46) and the rotor blade overlap data of Figure 47 are both arranged to show the effect of gas generator airflow variations. The fuselage length for this case is expressed as:

$$L_F = 22 + 0.5D + L_{GG} + L_{DV} + L_{PT}$$

where

L_F = fuselage length (ft)

D = main rotor diameter (ft)

L_{GG} = length of gas generator (ft)

L_{DV} = length of diverter valve (ft)

L_{PT} = length of power turbine (ft)

The percentage overlap of the tandem rotor blades can be expressed as:

$$\text{Percent overlap } X/D = (1 - L_R/D) \times 100$$

where

X = overlap

D = main rotor diameter

L_R = distance between rotor centers

Landing Gear - A fully retractable tricycle landing gear was employed on all aircraft in the study. Nose wheels were retracted forward into a bay beneath the cockpit area. The main landing gear of the propulsion-unloaded single rotor type was located in a stub or sponson projecting from the lower midsection of the fuselage; this assembly also provides space for fuel. In the case of the low-winged single rotor types, the sponson forms the inboard section of the wing and again provides space for fuel and main landing gear stowage after retraction. The sponson provides the structural attach points for the wing spars and, together with fuselage main frames, provides a wing carry-over structure.

All tandem rotor aircraft main landing gear assemblies are housed in a sponson running along the lower sides of the fuselage. The sponsons also provide space for fuel and, in addition, the aft sections contribute to the structural stability of the open-section aft fuselage in the cargo ramp area.

Flight Controls - Vertical and attitude control of all propulsion-unloaded compound aircraft is achieved throughout the flight regime by conventional helicopter control actions in terms of proper combinations of rotor blade collective or cyclic pitch inputs. Control of lift/propulsion-unloaded compound and composite aircraft is similarly achieved in the lower speed regimes of flight, with attitude and altitude control accomplished by conventional airplane cruise powerplant settings and flight surface actuation after conversion to high-speed cruise flight has been effected. In this flight mode, winged aircraft make use of aileron, elevator, and rudder control surfaces.

No specific design definition of aircraft flight control systems has been made for the purposes of this study. The required estimates of weight have been based on previous design and study experience for each type of aircraft, assuming employment of conventional hydromechanical systems.

DRAG TRENDS

The aircraft drag data employed in this study was based on a review of the summary results of wind tunnel tests of transport helicopter models conducted by the Vertol Division of Boeing. The tests were run on vehicle models representing the aircraft size category considered in this study. The wind tunnel data reflected present-day drag levels and also indicated the areas where drag improvements were attainable. Review of the data indicated that the major items contributing to the total aircraft drag were the rotor hubs, fuselage, and landing gear. These items together accounted for approximately 70 percent of the total drag. To achieve improvements in aircraft performance and to remain compatible with the advancing state of the art, it was evident that certain vehicle drag reductions could and should be accomplished.

Use of a retractable landing gear provided a significant improvement by effecting an approximate 25-percent reduction in the overall drag level. The fuselage afterbody shape in the rear ramp area also has a significant effect on the drag level. This major effect of contour results from the amount of flow separation in the aft fuselage region associated with the type of ramp configuration and the afterbody contraction ratio. A reduction of 30 percent of the fuselage drag for current rear-loading helicopters was attainable with a design incorporating a lower contraction ratio, this being defined as

the amount of contraction per fuselage length required for contraction.

The rotor hub category generally accounts for not only the hub proper and hinges but also the rotor blade shanks outboard to the beginning of the blade airfoil section. Improvements in the overall hub drag level can be achieved by the use of proper fairings or by boundary-layer control. Such improvements can account for approximately 30- to 50-percent reduction in rotor hub drag from that of current aircraft. Review of this data together with a review of the aircraft design layouts made for the purposes of this study was conducted to define the equivalent flat plate drag trends for the three types of aircraft involved. The typical component drag breakdowns for each type and configuration of aircraft are summarized in Table V. The general influence of disc loading and aircraft gross weight on profile drag is presented in Figures 48, 49, and 50 for the three aircraft types.

Propulsion-Unloaded Aircraft

This aircraft type is a pure helicopter variant incorporating a cruise fan system to provide auxiliary propulsive force. The design configuration incorporates pods along the lower portion of the fuselage which provide space for fuel and the retracted landing gear. Fuselage contours have been revised from those of current models in an effort to reduce drag substantially. A comparison of tandem and single rotor configurations shows a difference of less than 2 percent in total area.

This data in Table V was used as part of the development of the trends presented in Figures 48, 49, and 50 as indicated by the symbols at a gross weight of 35,000 pounds and a disc loading of 8 pounds per square foot. The increase in hub drag due to decreases in disc loading and increases of aircraft gross weight is caused by the increasing size of the rotor assembly. Similar changes in aircraft drag occur due to increases in airframe component size with decreasing disc loading and increasing gross weight.

Lift/Propulsion-Unloaded Aircraft

This aircraft type has the fuel and landing gear stowage pods integrated with the inboard wing structure in the low-wing single rotor case. For the high-wing tandem aircraft the pod configuration is similar to the propulsion-unloaded cases.

TABLE V

COMPARISON OF FLAT PLATE AREA - $f_e(ft^2)$

(GROSS WEIGHT 35,000 POUNDS, DISC LOADING 8 POUNDS PER SQUARE FOOT)

COMPONENT	PROPULSION-UNLOADED AIRCRAFT		LIFT/PROPULSION-UNLOADED AIRCRAFT		COMPOSITE AIRCRAFT	
	TANDEM	SINGLE	TANDEM	SINGLE	TANDEM	SINGLE
FUSELAGE	5.6	6.0	5.6	6.0	5.6	6.0
AFT PYLON	2.0	-	2.0	-	2.0	-
FORWARD PYLON	1.5	-	1.5	-	1.5	-
MAIN ROTOR PYLON	-	1.5	-	1.5	-	1.5
HORIZONTAL TAIL	1.4	-	1.4	-	1.4	-
FUEL PODS	1.6	1.5	1.0	1.5	1.0	1.5
ENGINE PODS	1.0	0.9	1.0	0.9	1.0	0.9
TAIL (PYLON AND STABILIZER)	-	0.8	-	0.8	-	0.8
HUBS AND SHANKS	9.0	-	6.4	-	-	-
HUBS	-	-	-	-	2.5	-
MAIN ROTOR HUB	-	11.0	-	8.0	-	3.2
TAIL ROTOR DRIVE AND ROTOR HUB	-	1.0	-	1.0	-	1.0
ROUGHNESS AND LEAKAGE	0.9	0.8	0.8	0.8	0.8	0.8
PROTUBERANCES	1.0	1.0	1.0	1.0	1.0	1.0
COOLING AIR MOMENTUM	0.9	-	-	-	-	-
COOLING MOMENTUM	-	0.9	0.9	0.9	0.9	0.9
TOTAL AREA (FT SQ)	24.9	25.4	21.6	22.4	17.7	17.6

Fuselage contouring for minimum drag characteristics commensurate with the study design requirements has again been effected. The wing contribution to aircraft drag is not included. Wing drag was calculated in the standard manner by means of the equation:

$$C_{d_{wing}} = 0.01 + \frac{C_\ell^2}{\pi eAR}$$

where

AR (aspect ratio) = 6.0

e (Oswald efficiency factor) = 0.8

C_ℓ = operating lift coefficient.

<u>Composite Aircraft</u>

This aircraft type has fuel and landing gear pods similar to the lift/propulsion-unloaded compound types. Use of fairings around the stowed rotor hubs and blades for high-speed airplane-type forward flight yields a significant net decrease in drag characteristics with respect to the other aircraft types. Again, the component drag breakdowns do not include the wing drag increment. Wing drag was accounted for in the same manner indicated previously.

<u>BASIC PROPULSION SYSTEM REQUIREMENTS</u>

<u>Typical Aircraft Power and Thrust Requirements</u>

<u>Hover Power Requirements</u> - The installed power requirement for aircraft employing all propulsion system types except independent system 3 is defined as the sea level standard value commensurate with power availability for hover out of ground effect at a 6000-foot, 95°F condition. An engine power lapse rate between these conditions of 0.78 was used. For system 3 the hover criterion defines the installed power of the lifting engines only. Study ground rules required that all aircraft be capable, at design gross weight, of hovering at the 6000-foot altitude, 95°F temperature condition while carrying the useful load necessary to meet the design mission requirements.

To provide a means of determining installed power requirements and propulsion component sizing in cases governed by the above hover condition, a computation method was selected and parametric data was defined to cover the required matrix of configuration and sizing cases. The installed power requirements were defined in terms of gas generator design airflow at sea level standard conditions for each combination of aircraft type and propulsion system. For this purpose, gas generator sizing data sheets, similar to the sample shown in Figure 51, were employed for the various aircraft/propulsion combinations. Computations were performed for these combinations across the aircraft gross weight range of interest using the study disc loadings. A sample of the resulting curves of gas generator airflow requirements against gross weight and disc loading is shown in Figure 52. In general, this plot shows the expected trends of increasing airflow/power requirements with higher gross weights and disc loadings. The method of computation involved in all cases is described below.

The specification of input parameters (aircraft type, propulsion configuration, disc loading, and assumed gross weight) permits estimation of hover power required at the rotors at a 6000-foot altitude, 95°F temperature. This is accomplished by the use of lifting rotor performance nomographs employing parameters of disc loading, estimated download in percent of thrust, rotor blade loading and rotor overlap where applicable, thus yielding a required rotor power loading. The specific means of determining rotor power loading is shown in Figures 53 and 54 for sea level standard and 6000-foot 95°F conditions respectively. As seen from the example in these figures, an estimate of download due to rotor wash is required in the calculations. For a specific disc loading and gross weight, a rotor radius is defined, and a percent download value is obtained from curves of download versus rotor radius. Figure 55 is a sample of this type curve. Use of the required power loading curves is as follows.

With disc loading and percent download values, enter Figure 53 and move parallel to the sloping lines as indicated by the dashed line of the example. Upon reaching the abscissa of the plot, follow the constant disc loading value to the blade area loading line employed for the aircraft being considered. At this intersection the required power loading (gross weight/rotor horsepower) is defined for a case of zero rotor disc overlap. An overlap condition (also shown in Figure 55) results in a lifting capability decrease as indicated in Figures 53 and 54.

Proper drive system efficiencies and other drive losses were applied in each case, based on system definitions shown in another section of this report, to refer this rotor power requirement to the engines. In addition, consideration was given to the characteristic engine power lapse rate with increasing altitude and temperature to define the sea level standard day installed power requirement for the gas generators. This power was then translated into terms of gas generator airflow requirements making use of the powerplant data generated for the study. Scaling of components as to size and weight was also based on the design airflow using the previously generated scaling data.

It should be noted that all components of the rotor drive systems were designed to accept the full sea level standard day power output of the powerplants. An airflow correction for required aircraft accessory drive shaft horsepower was made directly at the gas generators, regardless of the variations of the actual systems involved, since it was determined that the sensitivity of this assumption in the overall computation was extremely low.

Power Requirements in Forward Flight - The power requirements in forward flight for the aircraft of this study were defined by the use of generalized performance charts. This data was used to provide the capability of covering the large spectrum of rotor definitions in terms of such items as rotor radius, solidity, and lift requirements. Specific power requirements at sea level standard conditions for the optimum aircraft generated during the study, presented in a later section of this report (see pages 374 through 379), were separated into the three basic aircraft types considered: propulsion-unloaded, lift/propulsion-unloaded, and composite. This data is presented in a generalized format to illustrate the basic trends in terms of the ratio of power required to installed power, and thrust required to thrust at maximum speed (see Figures 56, 57, and 58).

Typical requirements for a propulsion-unloaded aircraft are presented in Figure 56. The hover requirement at sea level standard conditions is approximately 75 percent of the installed power for the design gross weight. The aircraft flies basically as a helicopter up to a speed where rotor limitations necessitate the use of the cruise fans to generate additional propulsive force. The speed at which the cruise fans are initially used is approximately 175 knots. At this speed the rotor power requirement is 65 percent of the installed value.

With a higher forward speed the rotor power requirement slowly decreases as cruise fan thrust is increased to 100 percent at maximum speed. The minimum power requirement (35 percent) occurs at approximately 80 knots. This minimum is slightly higher than that of the other aircraft types since, in the case of the propulsion-unloaded type, the rotor blade area is determined by the forward flight requirements and is greater than that necessary to meet hover requirements. In the case of the lift/propulsion-unloaded and composite aircraft, the blade area is defined by the hover requirement.

The typical requirements for a lift/propulsion-unloaded aircraft are presented in Figure 57. At sea level standard conditions, 65 percent of the installed power is required to hover. This value is slightly lower than the propulsion-unloaded configuration as a result of a lower download in the particular case illustrated. As in the propulsion-unloaded aircraft, the cruise fans are used to provide the auxiliary thrust required when the rotor limitations are reached. This point occurs at approximately 150 knots, a considerably lower speed than in the propulsion-unloaded aircraft case. This is caused by the additional drag of a wing, which imposes a large increase in the propulsion requirement. For example, a wing similar in geometry to those used in this study and operating at a lift coefficient of 1.0 with an area of 300 square feet yields an effective flat plate area increment (Δf_e) of approximately 25 square feet, resulting in the increased power requirements. This additional wing drag also results in an increase in the speed at which the minimum power required point occurs. With a wing the rotor lift requirement is greatly reduced in forward flight. Therefore, the rotor rpm is reduced to minimize rotor power requirements. At maximum forward speed the rotor rpm is reduced to give a value of advance ratio (μ) of 0.8. This reduction results in rotor operation near the autorotational state at high speed.

The composite aircraft requirements are presented in Figure 58. Rotor power required to hover at sea level standard is 73 percent of installed power. This value is slightly greater than that of the lift/propulsion-unloaded aircraft case as a result of a download increase for the particular aircraft shown as the sample. The rotor power required decreases until the transition speed of 125 knots is reached, at which point the lift is entirely on the wing. The result is a large amount of rotor unloading at low speeds, thereby requiring the use of cruise fan thrust at 80 knots. The

thrust requirement at the transition speed is approximately 70 precent of the thrust at maximum speed; it decreases rapidly to a 60 percent value at 160 knots. At the transition speed the rotor power required is about 12 percent of the installed value. Completion of the stopping, folding, and stowing process for the rotors eliminates the drag of stowage bay fairings and drag of the rotor. For speeds greater than that of transition, all lift is provided by the wing and all propulsion thrust is provided by the cruise fans in the manner of conventional airplane flight.

Propulsion System Design Requirements

Basic Systems Concepts - The cruise fan propulsion systems studied are of two basic types: integrated and independent. Those designated in this report as 1a, 1b, 2a, and 2b are integrated systems. System 3 is the independent type. The fundamental systems concepts are illustrated in Figure 59.

The differences between the two systems are the number of power sources and their use. The integrated system incorporates a single power source for both rotor lift and fan thrust. A switching and power management arrangement is incorporated to control direction and relative magnitudes of available power, depending on the demands of the two loads through the flight spectrum of the aircraft. Except for forward speed effects on power available at the source, or arbitrary mechanical limits in the system, total power available is a constant value throughout the flight spectrum. Determination of the installed power at the source can be based on a requirement at any design flight condition of the vehicle; however, a choice fixes the available power for all other conditions.

The independent propulsion system uses two separate power sources: one for the rotor system and one for the fan system. No switching device is incorporated, and flight shutdown of either portion, if required by the aircraft type, is accomplished at the applicable power source. The source-power design values are determined individually for each independent subsystem by different governing flight conditions. In this study the power requirement of the vertical lift subsystem is based on a design hover condition. The design power of the cruise fan subsystem is based on a specified vehicle high-speed requirement which varies with the aircraft type.

Comparison of relative worth of the two systems in an aircraft is a function of the type of machine and its relative power requirements at, for example, hover and high-speed conditions. If worth is measured only on the basis of total amount of installed power required, the integrated system is always better, since it requires less installed power. In general, the installed power requirement for an independent system varies between a theoretical 100 percent and a maximum of 200 percent of the installed power required for an integrated system. The former figure would apply in a hypothetical low-speed machine where the ratio of hover/cruise power required was infinitely large or to a hypothetical high-speed machine where the above ratio was extremely low. The 200-percent figure would apply where the hover/cruise power requirement ratio was one; that is, where required hover and cruise powers were identical. In the case of a large power requirement mismatch where the hover power requirement predominates, the integrated system may be penalized by inefficient part-power operation in cruising flight. If the cruise power requirement is dominant, the integrated system hover inefficiency, in terms of power matching, may be practically aided by torque-limiting the lift system. In the case of multiengined installations, reserve thrust for emergency engine-out conditions can then be made available. However, in normal instances where a severe power mismatch problem is not present, the integrated system has the advantage of minimizing installed power required. In addition, minimizing the number of power sources is important from practical design installation, weight, reliability, and system cost aspects. To obtain the full potential benefits of an integrated system, however, the power switching and management functions required must be implemented in the simplest possible manner.

Typical variations of rotor power and cruise fan thrust requirements against sea level flight speeds for the three types of aircraft in this study have been shown previously in Figures 56, 57, and 58. In considering the basic requirements of propulsion systems, it is desirable to review these variations of demand on total installed power to determine in each case where, to what degree, and in what relative proportions it must be employed.

<u>Integrated Propulsion Systems</u> - The propulsion-unloaded aircraft type requires rotor power throughout its operating speed range. The peak standard-day requirement, corresponding in a typical case to about 75 percent of installed 6000-foot, 95°F hover power, occurs at the hover condition. Power demand from

the rotor drops off rapidly with increasing speed in characteristic helicopter fashion to about 30 percent of installed hover power at about 40 percent of maximum speed. The rotor requirement then increases until at the start of the cruise flight regime (about 85 percent of high speed) it absorbs slightly less than the standard-day hover requirement. At this point, cruise fan power and thrust to unload the rotor are applied on a rapidly increasing basis with speed, with peak fan powers occurring at maximum speed. This fan thrust, applied over the upper 15 percent of the aircraft speed range, reduces rotor power requirements so that at maximum speed the division of installed power is approximately 60 percent to the rotor and 40 percent to the cruise fans.

The lift/propulsion-unloaded aircraft type requires rotor power in generally decreasing amounts from a peak value at hover as flight speed is increased. At 50 to 65 percent of maximum speed, the rotor power requirement at standard conditions has reduced to about 25 to 35 percent of installed (6000-foot, 95°F) hover power, and further reduces to zero at maximum speed. Cruise fan thrust is applied at approximately two-thirds maximum speed. The fan thrust and power required increase over the upper third of the aircraft speed regime to the point where the total installed power is absorbed by the fans at maximum speed.

Composite aircraft require rotor power at speeds below transition. The peak rotor requirement occurs at hover; it continually decreases with speed to a minimum of about 10 to 15 percent of the amount installed for a 6000-foot, 95°F hover condition as the aircraft reaches the 125-knot transition speed. Since the rotor stopping and stowing operation is effected in this speed range, required rotor power then goes at zero. Power is required by the cruise fans at well below this point, however, or around 80 knots. In the case of aircraft with integrated propulsion systems, about 60 percent of available thrust is needed at transition speed due to a high drag condition of the aircraft. At speeds greater than transition, the fan power requirements decrease slightly and then increase in conventional airplane fashion until all the installed power is absorbed by the fans at the aircraft maximum speed.

Independent Propulsion System - Rotor power requirements for the three aircraft types generally exhibit the same trends previously discussed, and the lift system installed power is based on a 6000-foot, 95°F hover requirement. Since the cruise

fan systems are independent of the rotor systems, installed fan power is separately determined from a vehicle high-speed requirement. In propulsion-unloaded aircraft the rotor power requirement tends to decrease from the high-speed value noted in the integrated system case above as vehicle speed is further increased. Rotor power requirements of lift/propulsion-unloaded aircraft exhibit the same trend in going to zero at maximum speed; however, fan thrust is employed over the upper 45 percent of the vehicle speed range since the maximum speed of these aircraft has been increased to 270 knots. The same is true of composite aircraft with the high speed increased to 350 knots.

Integrated Systems Requirements - The above review of aircraft power requirement trends assists in understanding the following basic requirements for integrated cruise fan propulsion systems. For potential application to all aircraft types, these systems must allow:

1. Application of full installed power to the rotors, preferably with no power to the fan, and minimum transmission losses.

2. Application of low percentages of installed power (down to idle) to the rotors alone, with no fan power applied, and capability for smoothly modulated power changes in both directions.

3. Application of smoothly modulated power to the fans from idle to the maximum installed power without absorption of power by the rotors. This requirement is for a composite aircraft.

4. Use of any division of power between fans and rotors from idle to maximum installed power with minimum system losses, and a precise means of control of the power division by the pilot.

5. Pilot-selectable constant speed (rpm) control of the rotor system, and constant speed control or topping speed control of the fan system, depending on the use of shaft- or gas-driven systems.

6. Selected reduction of system speed to approximately 60 percent of maximum operating rpm without severely penalizing fan performance in shaft systems.

Integrated systems must in general have the following design provisions:

1. A means of totally unloading power absorption of either rotors or fans, that is, a switching system

2. A means of modulating power in each leg of the system, that is, a relative power management system

3. A means of varying fan power load in shaft-driven systems

4. A means of limiting fan speed in gas-driven systems

5. A means (in multiengine aircraft) of taking an engine off the line without motoring by the drive system

6. A method of ground or air starting any engine of a multi-engine system

7. A method of driving aircraft accessories at all times, including partial or complete power-off situations

8. Depending on the particular aircraft installation, it may be desirable to provide symmetrical fan thrust under engine-out conditions of multiengine, multifan arrangements

Independent Systems Requirements - The basic requirements of the lift system portion of an independent cruise fan propulsion system are:

1. Application of full installed lift engine power to the rotors with minimum transmission losses must be allowed, and provision for reduction of power to idle with smooth variation of part power must be made.

2. The lift system must be capable of shutdown and air start when used in composite aircraft.

3. A portion of the lift system must drive all aircraft accessories.

4. A means of disconnect must be provided to take an engine off the line without motoring by the rotor drive system in multiengine aircraft.

5. Pilot-selectable constant speed (rpm) control of the system, with the speed range of the engine compatible with rotor speed reduction requirements, must be provided.

The basic requirements of the fan part of the independent system are:

1. Smoothly variable fan thrust availability from idle to maximum thrust conditions

2. Ground and air start provisions

3. Aircraft accessories must be driven by this subsystem in composite aircraft types

4. Cruise fan power interconnection may be desirable depending on engine-out thrust asymmetry conditions in a particular aircraft installation

PROPULSION SYSTEMS DESIGN DESCRIPTION

As noted in the introduction, five propulsion systems were to be integrated with each type of aircraft studied. Figures 60 through 69 show design schematics covering the range of these systems. The schematic drawings show basic arrangements of engines, engine-mounted accessories, subsystems required for engine operation, engine-fan and engine-transmission coupling means, switching and clutching devices, aircraft accessory devices, and rotor-drive transmissions. In addition, the location of power management devices is indicated. A general description of each system is given below.

Integrated Gas-Coupled Systems (1a and 1b)

These systems are shown in Figures 60 through 63 for single and tandem rotor aircraft of all types. The only difference between the 1a and 1b systems is in the method of cruise fan gas drive, one being tip driven and the other hub driven. Twin gas generators are connected by gas ducting to both cruise fans and to a common power turbine that drives the rotor transmission in a manner allowing use of these gas generators in all flight conditions. The system is arranged so that either or both gas generators can provide power to the rotors, but each gas generator can provide power only to the cruise fan on its side. Diverter valves are used at each gas generator exhaust to act as full shutoffs in either fan or rotor drive

directions (use of the rotor-drive shutoff is dependent on the type of aircraft considered) or to allow gas flow in both fan and rotor directions at the same time. A variable gas-admission device is indicated on the schematic at both the cruise fan entry and the common power turbine entry to provide means of controlling power flow between the two paths (power management) when both systems operate on a power sharing basis. The "subgroups not shown" block noted on the gas generators calls out the standard powerplant subsystems required in support of gas generator operation.

A remote accessory drive gearbox is indicated in the schematic which shows inputs going through overrunning clutches from an auxiliary powerplant, from both gas generators, and also from the rotor transmission. This remote accessory drive gearbox permits aircraft accessories to be driven during all modes of aircraft operation, including cases of remote base self-sufficiency, single engine operation, or autorotation.

The rotor-drive transmission is shown for both single and tandem rotor aircraft, and main gearboxes, cooling, and braking provisions are indicated. Since the operating speeds of the common power turbine are considerably higher than those selected for the main rotor gearbox drive shafts, a power turbine reduction gearbox is incorporated in the system with an overrunning clutch on the turbine side to eliminate turbine motoring during autorotation.

The gear arrangement chosen for this turbine reduction gearbox is of the star type; that is, a sun gear is connected to the common power turbine shaft, a set of fixed planets is mounted to the housing, and the output is taken through the rotating ring gear. This type of system is superior to a pure planetary reduction where very high-speed operation (20,000 rpm) is required, since the large centrifugal load on the planet bearings is eliminated. Another feature of the star arrangement is that greater flexibility of ratios is possible since a true planetary system cannot achieve a reduction ratio of less than approximately 2.24 to 1. The details of the rotor-drive transmission for propulsion systems 1a and 1b are shown in Figures 70 and 71 for single and tandem rotor aircraft, respectively. Values are shown for shaft speeds, gear ratios, and efficiencies; gear configurations are also indicated. These gear configurations were reviewed for applicability over the expected scaling ranges of rotor diameter and speeds and for range of power turbine input speeds. The arrangements shown

are suitable over the range considered.

The engine gearboxes are configured using a single spiral bevel gear set. The coupling or combining gearbox used for scaling in all cases consists of two spiral bevel pinions driving a third gear. An alternative configuration of this gearbox which could be used in cases where the engines are very close to the synchronizing shaft might consist of spur reduction systems where two pinions mesh with a large driven gear.

Hydromechanical clutch systems to permit stopping and starting the rotor are used in composite aircraft. They employ hydraulic couplings similar to those used in automobile automatic transmissions. This coupling is used to start the synchronizing shaft and accelerate it to near full rpm. When a preset speed is achieved, a friction or jaw type of clutch is used to complete the power path so that full system power may be transmitted.

The main rotor transmissions for single and tandem rotor aircraft consist of a primary spiral bevel set employed to change direction and to take a moderate reduction, and a two-stage planetary set. The arrangement takes a larger reduction in the first stage and a smaller reduction in the final planetary stage. This gear selection was made because a large reduction in the first stage allows the planet carrier to turn slower; this places more moderate centrifugal loads on the first-stage planet bearings and allows a more favorable sizing for weight. Use of the smaller reduction in the second planetary stage permits use of a larger sun gear accommodating a larger number of planets, thus providing a very effective way of minimizing weight because of the high torque levels at the final output rpm.

In the case of the antitorque gear transmission for single rotor aircraft, a simple spiral bevel gear set is used for the intermediate transmission and a spiral bevel set with appropriate reduction is used at the tail rotor.

Further definition of the major gearboxes for scaling purposes is given by the following equations:

<u>Power Turbine Gearbox</u>

$$D_{PTB} = 0.569\ (SHP_{PT})^{1/3}$$

where

D_{PTB} = diameter of gearbox over the PTB ring gear (in.)

SHP_{PT} = shaft horsepower at power turbine (at constant output shaft rpm)

Rotor Gearbox

$$D_{RGB} = 0.374 \ (D \times SHP_{RGB})^{1/3}$$

$$H_{RGB} = 1.8 \ D_{RGB}$$

where

D_{RGB} = diameter of rotor gearbox (in.)

D = rotor diameter (ft)

SHP_{RGB} = shaft horsepower per rotor gearbox

H_{RGB} = height of rotor gearbox (in.)

This data is based on a review of actual hardware and study designs of gearboxes. In the single rotor case, the sizing for design of the tail rotor gearboxes was not considered necessary for the purposes of the study; therefore, only data sufficient for weight estimation was defined.

Integrated Shaft-Coupled Systems (2a and 2b)

These systems are defined in Figures 64 through 67 which cover all aircraft types in single and tandem rotor configurations. The essential difference in these systems from the 1a and 1b gas-coupled arrangements is the linking of prime movers and loads by mechanical power drives throughout. The two shaft drive systems differ in that system 2a employs shaft turbine powerplants located remotely and inboard from the cruise fans, and system 2b locates the shaft turbines coaxially and together with the fan assemblies in an outboard location. As before, the systems are integrated so the twin powerplants can be employed throughout the aircraft flight regime. These shaft systems have the feature not present in the gas-coupled arrangements that permits either or both powerplants to drive the rotors, and also allows either or both to drive both cruise

fans because of the cross-shafting arrangement.

In both shaft-coupled systems, the free power turbine of each engine drives into a cross-shaft gearbox through an overrunning clutch, which enables disconnection of an inoperative engine from the system. At this point, the shaft arrangements provide for splitting power between twin cruise fan drive systems and the rotor drive-transmission in proportion to relative fan and rotor load demand. Cruise fan decoupling clutches allow disconnection of any fan load during the helicopter mode of flight. Depending on the selected vehicle type, the system may incorporate the capability of disconnecting the rotor drive-transmission from the cross-shaft system by means of a decoupling clutch at the output of the coupling gearbox. This clutch is incorporated in the system when used in a composite type of aircraft where no power is required by the stowed rotors during cruise flight. In the case of application to the compound types of aircraft no clutch is required. Between the end points of rotor or fan load shut-off by disconnection, power demand between the fans and rotor system may be changed by varying settings of rotor blade pitch in combination with a load control device incorporated in the fan assemblies. For this study, variable inlet guide vanes are used for this control. As in the gas-coupled system schematics, the powerplant subsystems required are indicated simply as blocks to denote their required presence. The aircraft accessory drive gearbox in the shaft-driven systems can be kept operative in all flight modes as well as for ground operation from two drive inputs using overrunning clutches; one from the coupling gearbox (since the cross-shaft is running regardless of engine shut-down condition), and the other from the auxiliary powerplant.

In the remote shaft turbine arrangement of system 2a, two bevel gearboxes per side are required to route power from shaft turbines to fans. The axially driven concentric fan arrangement of system 2b also involves the use of either one or two gearboxes based on the rpm relationship between the power turbine and the fan.

The rotor-drive transmission details for the shaft-coupled systems are shown in Figures 72 and 73 for tandem and single rotor aircraft. Again, values of shaft speeds, gear ratios, and drive efficiencies assumed for the study have been indicated either as constants or set up in scaling form after review of the range of major input values to be considered

and after review of actual rotorcraft transmissions and transmission design studies. Scaling of sizes of major transmission components is shown by the following equations.

Rotor Gearbox

$$D_{RGB} = 0.374\ (D \times SHP_{RGB})^{1/3}$$

$$H_{RGB} = 1.8\ D_{RGB}$$

where

D_{RGB} = diameter of rotor gearbox (in.)

D = rotor diameter (ft)

SHP_{RGB} = shaft horsepower per rotor gearbox

H_{RGB} = height of rotor gearbox (in.)

Combining Gearbox

$$D_{CB} = 0.50\ (SHP_{CB})^{1/3}$$
(at 7155 rpm output shaft speed)

where

D_{CB} = diameter of combining gearbox

SHP_{CB} = shaft horsepower at combining gearbox

Decoupling Clutch

$$D_{CL} = 0.85\ (SHP_{CB})^{1/3}$$

$$L_{CL} = 1.5\ D_{CL}$$

where

D_{CL} = diameter of decoupling clutch (in.)

SHP_{CB} = shaft horsepower at combining gearbox

L_{CL} = length of decoupling clutch (in.)

Independent Lift and Cruise Propulsion System (3)

The configurations of the independent systems employing concentric cruise powerplants and fans are depicted in Figures 68 and 69 and cover all aircraft types in both single and tandem rotor configurations. The fundamental difference from the previously discussed integrated systems is that separate or independent twin sets of powerplants are used for rotor and fan-powered flight, with no power connection between the two subsystems. Therefore, a total of four gas generators is required. Shutdown of either twin-powerplant subsystem can be effected when not required in a particular flight regime. In addition, although either or both of the rotor drive shaft turbines can supply power to the rotor, the cruise fan systems are depicted as independent of each other, since no power cross-shaft connection is made and unsymmetrical thrust in cruising flight would result from the fan system should either cruise powerplant be shut down. If found desirable from an aircraft control or performance point of view, a modification could be made to this system to provide fan thrust symmetry in the event of powerplant shut-down by employing a power cross-shaft between the cruise fan assemblies.

The rotor drive subsystem consists of two gas-coupled shaft turbine engines that provide power to the main rotor gearboxes through overrunning clutches, allowing engines to drop off the line with shut-down; a bevel gearbox connection to the cross-shaft; a coupling or combining gearbox linking the cross-shafting to the rotor gearbox drive shaft; and conventional rotor drive gearboxes. No decoupling clutches are required, since the whole subsystem can be made inoperative where appropriate to the type of aircraft and flight condition. Inlet closure doors are indicated as desirable for the high-speed composite aircraft type, so windmilling drag is not incurred.

The separate cruise fan assemblies are of the conventional concentric front-fan type. Since the fans are separate items, the options of fan shutdown or idling in the low-speed flight regime, or provision of fan thrust to aid rotor lift by tilting or slipstream deflection, are present. However, if lift were to be augmented by fan thrust, an intershaft system between fans to maintain thrust symmetry in the event of engine failure might be necessary. A discussion of the design installation feasibility of fan-augmented hover thrust is included in a later section (see page 184). The aircraft accessory gearbox is driven through overrunning clutches from both the coupling

gearbox in the rotor-drive system and the aircraft auxiliary powerplant when used in compound aircraft types. When the system is employed with composite aircraft and the rotors (and rotor-drive system) are stopped, additional inputs to the accessory box through overrunning clutches are required from each cruise fan drive assembly in an arrangement to cater to a case of engine shutdown in cruise flight.

Since four gas generators are required in the independent system, the schematics indicate that some of the powerplant system subgroups are required separately for each gas generator to make a total operative aircraft system.

Figures 72 and 73 give the details of rotor transmission drive and cross-shaft gearbox and shafting for this system, which are the same as in systems 2a and 2b except that the decoupling clutch is omitted, and the bevel gearbox at the shaft turbine output (engine gearbox) in the rotor-drive subsystem is included.

It should be noted that the reason the engine gearbox is included in this system, and not in the integrated shaft-coupled system 2a discussed previously, is related to whether or not these elements are included in the powerplant scaling data used as a basis for the study. As in the other systems, shaft speeds, gear ratios and efficiencies, and gearbox configurations are defined for system 3 in these schematic drawings. The equations given previously apply for size scaling of main rotor gearboxes, and coupling or combining gearboxes. Size scaling of the engine bevel gearbox is given by the equation:

$$D_{EB} = 0.432 \ (SHP_{EB})^{1/3}$$

where

D_{EB} = diameter of engine gearbox (in.)

SHP_{EB} = shaft horsepower at engine gearbox

Like the other propulsion systems, no size scaling has been defined for the tail rotor drive systems of single rotor aircraft; however, sufficient design data has been set up to allow estimates of weights of these components as well as drive train efficiencies for power estimates.

SUPERCRITICAL SHAFT SYSTEMS

During the study of transmission systems, consideration was given to the potential application of supercritical shafting. It was recognized that design studies have shown substantial weight advantages for this type of shafting. Some studies have shown weight decreases of 30 to 50 percent with respect to currently used shaft systems and have indicated some problem areas in installation and handling. Review of application in this study raised serious questions as to the feasibility, within constraints of the work statement, of making proper design and weight tradeoffs for total transmission systems over the complete scaling range required. Review of scaling in one tandem rotor case indicated that use of a higher speed supercritical synchronizing shaft would still not eliminate the requirement for gearing between the shaft and power turbine, but would in some scaling cases require use of a three-stage planetary system in the rotor drive gearbox. In single rotor aircraft the main rotor gearbox drive shaft is relatively short; questions arose here as to supercritical shafting applicability on a total systems basis along with the relative application between single and tandem rotor aircraft. It was concluded that proper resolution of all the tradeoffs involved was beyond the scope of the present study. Since the selection of shafting types did not have any first-order effect on the relative evaluation of propulsion systems and since it is believed to have little effect on the absolute values of aircraft weights, it was decided not to employ supercritical shafting in the transmission systems.

AIRCRAFT ACCESSORY DRIVES - DESIGN DATA

The assumption was made for all types and sizes of aircraft covered by the study that a nominal 60 auxiliary horsepower would be used as a design value for the aircraft accessory drive system. No attempt was made to assess the types of drives to the accessory drive box, their sizes, or the size and number of pads on the gearbox; however, the assumption was made that wherever design installation factors allowed, the systems would be mechanical. The weight of the auxiliary powerplant installation was held constant as part of the fixed equipment group in the study ground rules.

BASIS OF WEIGHT ESTIMATE

Following is a discussion of the methods used to derive the weights for each group shown on the weight summary sheet. Adjustment for advanced technology is made as a separate entry and is discussed in the section titled "1970 State of the Art" on page 206.

Rotor Group

The rotor group weight was derived by using the standard Boeing weight trend. The weights obtained from this trend represent a standard steel spar, aluminum-covered blade with a steel hub. The trend and the parameters defining the weight are as follows:

$$W_R = 14.2\ K^{0.67}\ (1.4)$$

$$K = r^{0.25}\left(\frac{HP}{100}\right)^{0.5}\left(\frac{V_t}{100}\right)\left(\frac{R_{bc}}{10}\right)\ K_d$$

where

- r = distance from axis of rotation to blade attachment point using 0.085R (ft)
- HP = transmission limit horsepower (using 0.6 HP for tandem rotor aircraft)
- V_t = 1.2 x hover tip speed (fps)
- R = rotor radius (ft)
- b = number of blades per rotor
- c = blade chord (ft)
- K_d = static droop penalty factor (ignored if less than 1.0)

  $\frac{R^{1.6}}{180c}$ for single rotor

  $\frac{R^{1.6}}{144c}$ for tandem rotor

  (1.4) = factor for blade folding on composite aircraft

W_R = weight of one rotor in tandem aircraft

K = correlating factor

Note: Rotor blade t/c is assumed constant at 15 percent at 0.25R

<u>Wing Group</u>

The Boeing-Vertol trend is used to determine the wing group weights. A weight penalty for aircraft using tilting wings is included in the flight controls group. The following constants were used to simplify the wing weights:

AR = aspect ratio = 6.0

t/c = thickness/chord ratio = 15%

n = ultimate load factor = 4.5

The simplified wing weight trend is:

$$W_w = 220\left(\frac{W_G}{10^4} \times 1.8 \times \frac{S_w}{10^2}\right)^{0.658}$$

where

W_G = design gross weight (lb)

S_w = wing area (ft^2)

<u>Tail Group</u>

Horizontal tail weights are estimated on the basis of wing area and tail moment arm. A standard unit weight of 2.5 pounds per square foot for single rotor aircraft and 3.0 pounds per square foot for tandem rotor aircraft was used.

The equations used are:

$$\text{Single rotor } W_{HT} = \frac{S_w}{D} \times 15.2 \times 2.5$$

$$\text{Tandem rotor } W_{TG} = \frac{8.57 \quad S_w}{26.7 + (D-63)} \times 3.0$$

where

S_w = wing area (ft^2)

D = rotor diameter (ft)

W_{HT} = weight of horizontal tail (lb)

W_{TG} = weight of tail group (lb)

To obtain tail group weight for single rotor aircraft, it is necessary to add horizontal tail weight and tail rotor weight as follows:

$$W_{TG} = W_{HT} + W_{TR}$$

where

$$W_{TR} = 16.05\ K^{0.67}$$

$$K = (0.1R)^{0.25} \left(\frac{HP}{100}\right)^{0.5} V_t \ \frac{Rbc}{10}$$

R = tail rotor radius (ft)

HP = tail rotor transmission horsepower

V_t = tail rotor hover tip speed x 1.2 (fps)

b = number of blades

c = blade chord (ft)

Body Group

The body group weight comprises two main sections: basic structure and secondary structure. All configurations are designed around a constant payload compartment 7.5 by 6.5 by 30 feet. For weight estimation the Boeing standard helicopter body trend is used to derive the basic structure, and a constant weight of 1,850 pounds (same as for CH-47) is used for the secondary structure.

Blade fold fairings for the composite aircraft, main rotors only, are based on a standard length of 12.5 feet and specific

weight of 3.5 pounds per square foot which includes fairings and actuators. The weight is given by:

$$4.8 \times \text{blade chord} \times 12.5 \text{ ft} \times 3.5 \text{ lb/ft}^2$$

The following expression is used to derive the weight of the basic structure:

$$W_{BS} = 280\ K^{0.5}$$

$$K = \frac{W_X}{10^4} \times \frac{S_f}{10^3}\ (L_F + L_X)^{0.5}\ (\log V_D)\ \ n k_1\ k_2$$

where

W_{BS} = weight of basic structure (lb)

W_X = weight of fuselage and contents (lb)

S_f = fuselage wetted area (ft^2)

L_F = fuselage length (ft)

L_X = length of ramp well (ft)

V_D = limit dive speed (knot)

n = ultimate load factor = 4.5

k_1 = 1.0 for land-based aircraft

k_2 = load distribution factor = 0.2

The fuselage wetted area consists of the standard fuselage having a length of 56 feet plus increments of wetted area for fuel pods, rotor pods, tail boom, and vertical tail. The wetted areas for fuel pods and rotor pods vary with the configuration, and the tail boom and vertical tail wetted areas vary with rotor diameter.

Landing Gear

The landing gear weight is based on a standard 3 percent of design gross weight for the tricycle VTOL landing gear plus 1 percent of design gross weight for the retraction system.

Flight Controls

The flight control weight is derived by using the following expressions:

Single Rotor

$$80 + 0.01W_G + 26\left(\frac{W_G}{10^3}\right)^{0.41} + 0.15\ W_{MR} + 30\left(\frac{W_{MR}}{100}\right)^{0.84} + 0.25\ W_{TR}$$

The term 0.01 W_G is deleted for propulsion-unloaded aircraft.

Tandem Rotor

$$80 + 0.01\ W_G + 26\left(\frac{W_G}{10^3}\right)^{0.41} + 0.3\ W_R + 60\left(\frac{W_R}{100 \times 2}\right)^{0.84}$$

The term 0.01 W_G becomes 0.015 W_G for winged tandems to allow for tilting. Divide rotor weights by 1.4 (blade fold factor) for composite aircraft before using in the above expression with the exception of single composites in which the tail rotor does not fold.

Engine Section

The weight of the engine section includes nacelle and mounts for the inboard engines and pods, and pylons and mounts for the outboard engines and/or cruise fans.

The expression

$$7\left(\frac{HP}{100}\right)^{0.67}$$

was used for inboard engines and

$$(W_{CF} \times \eta_{CR})^{0.41}$$

for outboard engines and/or cruise fans

where

HP = total engine horsepower

W_{CF} = weight of cruise fans or engines (lb)

n_{CR} = crash load factor = 20

Engines and Cruise Fans

Engine and cruise fan weights are obtained from the curves presented in the propulsion section of this study.

Air Induction

Air induction weight is included in the engine section.

Exhaust System

A standard allowance of 33 pounds is made for this item for all propulsion systems except 2b.

Cooling and Lubrication

Cooling and lubrication weights are included with engine weights.

Fuel System

A fixed allowance of 30 pounds per engine is included for the fuel system and controls plus a variable weight based on the fuel load to cover tankage. The fuel tanks have 30-percent self-sealing capability against .30 caliber fire.

$$\text{Fuel system} = \text{number of engines} \times 30 + \left(0.3 \times \frac{\text{fuel weight}}{6.5}\right)$$

Engine Controls

The engine controls weight is estimated at 20 pounds per engine.

Starting System

The starting system weight is estimated at 30 pounds per engine.

Drive System

The drive system weight is obtained by analyzing the design

layout for each system on a box-by-box basis and by deriving an expression for each gearbox. The expressions are simplified so that the variables are at a minimum. Generally the box weights may be determined by using transmission power, rotor radius, and a constant. In the case of the gas-coupled systems, the gas generator airflow is required to obtain reduction gearbox weights.

The expressions used are as follows:

P_x = total transmission horsepower

R = rotor radius (ft)

W_a = total lift engine airflow (lb/sec)

L = distance between rotors (ft)

Single Rotor

Main box = $(10.1R + 110)\left(\frac{P_x}{10^3}\right)^{0.8}$

Intermediate box = $25.5\left(\frac{P_x}{10^3}\right)^{0.8}$

Tail rotor box = $25\left(\frac{P_x}{10^3}\right)$

Reduction box = $\left(\frac{P_x}{10^3}\right)^{0.8}\left(\frac{158}{\sqrt{\frac{W_a}{2}}} - 6.6\right)$

Coupling box = $61\left(\frac{P_x}{10^3}\right)^{0.8}$

Engine box = single curve for all bypass ratios resulting from equation for each bypass ratio. (See Figure 74.)

Decoupling clutch = $0.02\ P_x$

$$\text{Shafting} = 4.83\left(\frac{P_x}{10^3}\right) + 1.16R\left(\frac{P_x}{10^3}\right)^{0.5} + 1.28\left(\frac{P_x}{10^3}\right)^{0.67}$$

$$\left.\begin{array}{l}\text{Lubrication system}\\ \text{Rotor brake}\\ \text{Cross shafts}\\ \text{Accessory drive pads}\end{array}\right\} = 20\%\text{ of box weights}$$

Tandem Rotor

$$\text{Forward and rear boxes} = \left(\frac{Px}{10^3}\right)^{0.8}(11.56R + 125)$$

Reduction box = as for single rotor

Coupling box = as for single rotor

Engine boxes = as for single rotor

Decoupling clutch = as for single rotor

$$\text{Aft shaft} = 1.25\left[\frac{0.24R\left(\frac{0.6\ P_x}{100}\right)^{0.5} + 7\left(\frac{0.6\ P_x}{100}\right)^{0.67}}{\left(\frac{7155}{R \times 10^3}\right)^{0.65}}\right]$$

$$\text{Synchronizing shaft} = 0.77\ \frac{P_x}{100}^{0.5} \times (L - 2.0)$$

Lubrication system, etc. = as for single rotor

<u>Fixed Equipment</u>

The following items of fixed equipment are constant for all aircraft in the study.

	Weight (lb)
Auxiliary powerplant	100
Instruments and navigation	170

	Weight (lb)
Hydraulics and pneumatics	225
Electrics	550
Electronics	285
Armor	300
Furnishings and equipment	860
Airconditioning and deicing	180
Auxiliary gear	265
Total	2935

Advanced Technology

The airframe items from rotor group to engine section are totaled, and 5 percent of this figure is subtracted from the total as an advanced technology factor. In addition, 10 percent of the drive system weight is subtracted for this reason.

The reductions in weight for advanced technology are discussed further in the paragraph headed "1970 State of the Art" on page 206.

Fixed Useful Load

Three crew members	=	600 lb
Trapped liquids	=	70 lb
Engine oil	=	80 lb
Total	=	750 lb

Fuel

The fuel load determined by the aerodynamics group is consistent with the mission specified for the study.

Payload

A constant payload of 6000 pounds is specified for the study.

For this study the exhaust, engine controls, starting system, trapped liquids, and engine oils are assumed to be constant. These items would vary slightly with engine size and aircraft configuration, but the small changes do not justify detailed calculations.

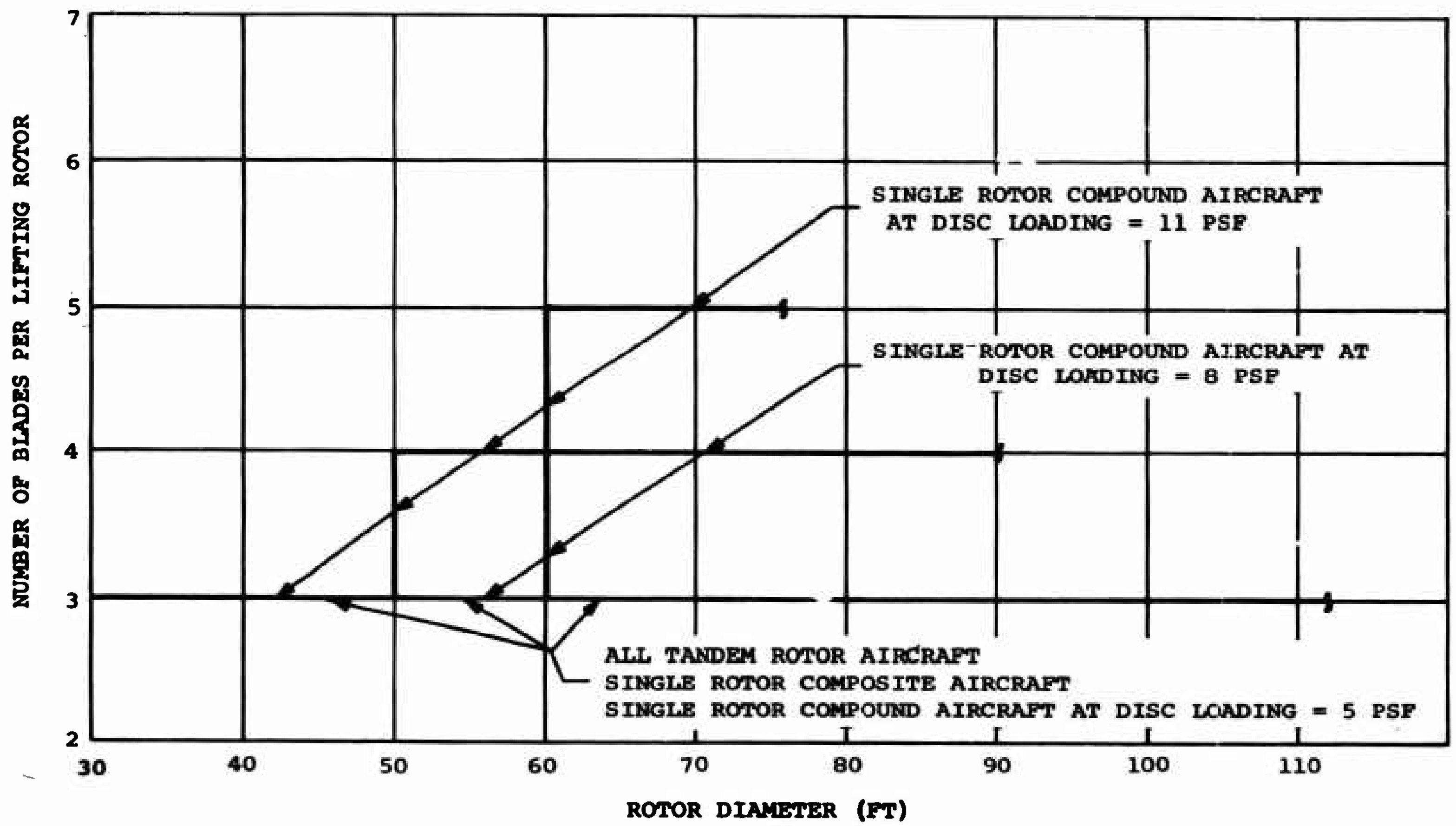


Figure 42. Number of Blades per Lifting Rotor vs. Rotor Diameter

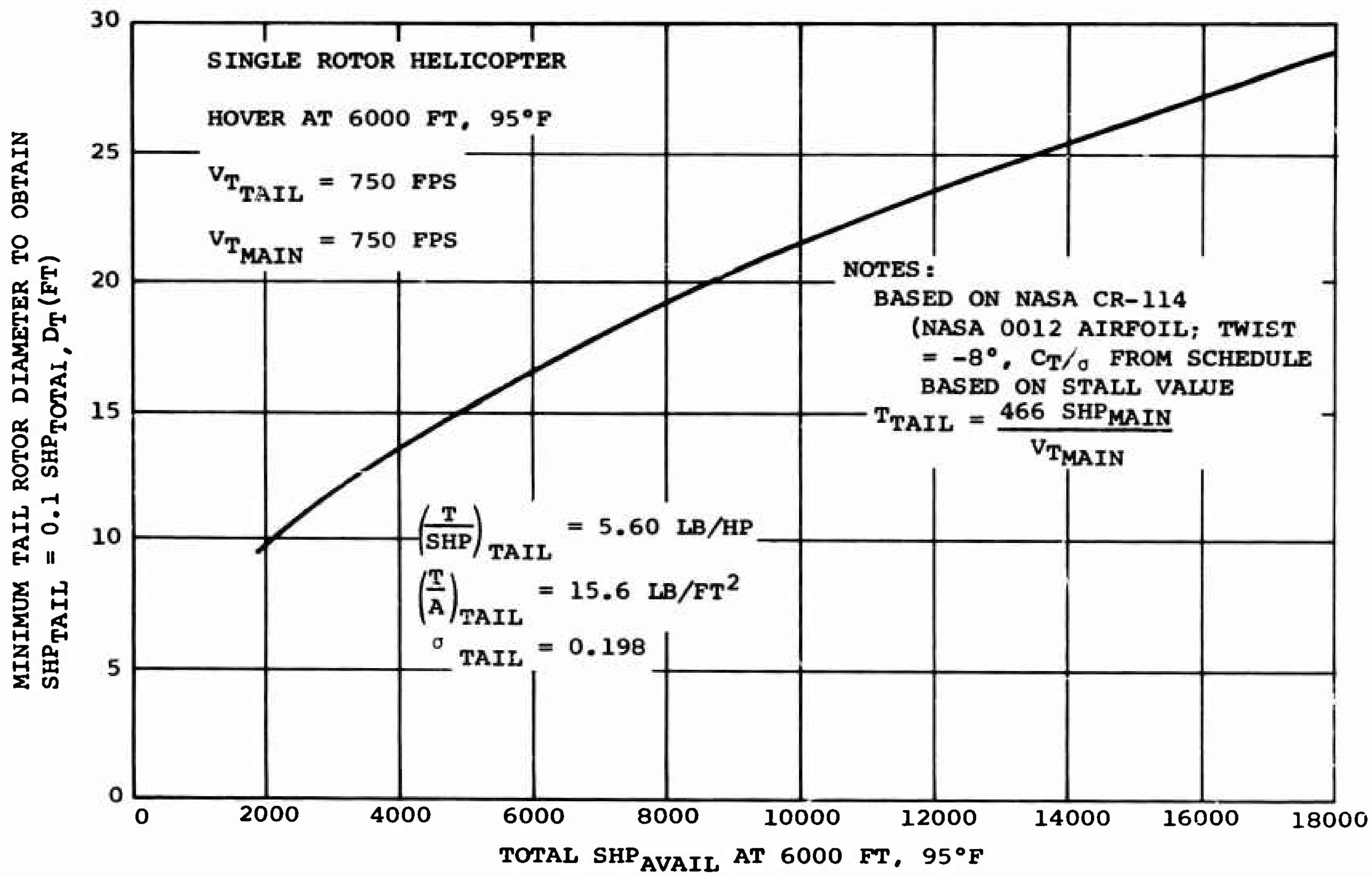


Figure 43. Preliminary Design Data - Tail Rotor Size

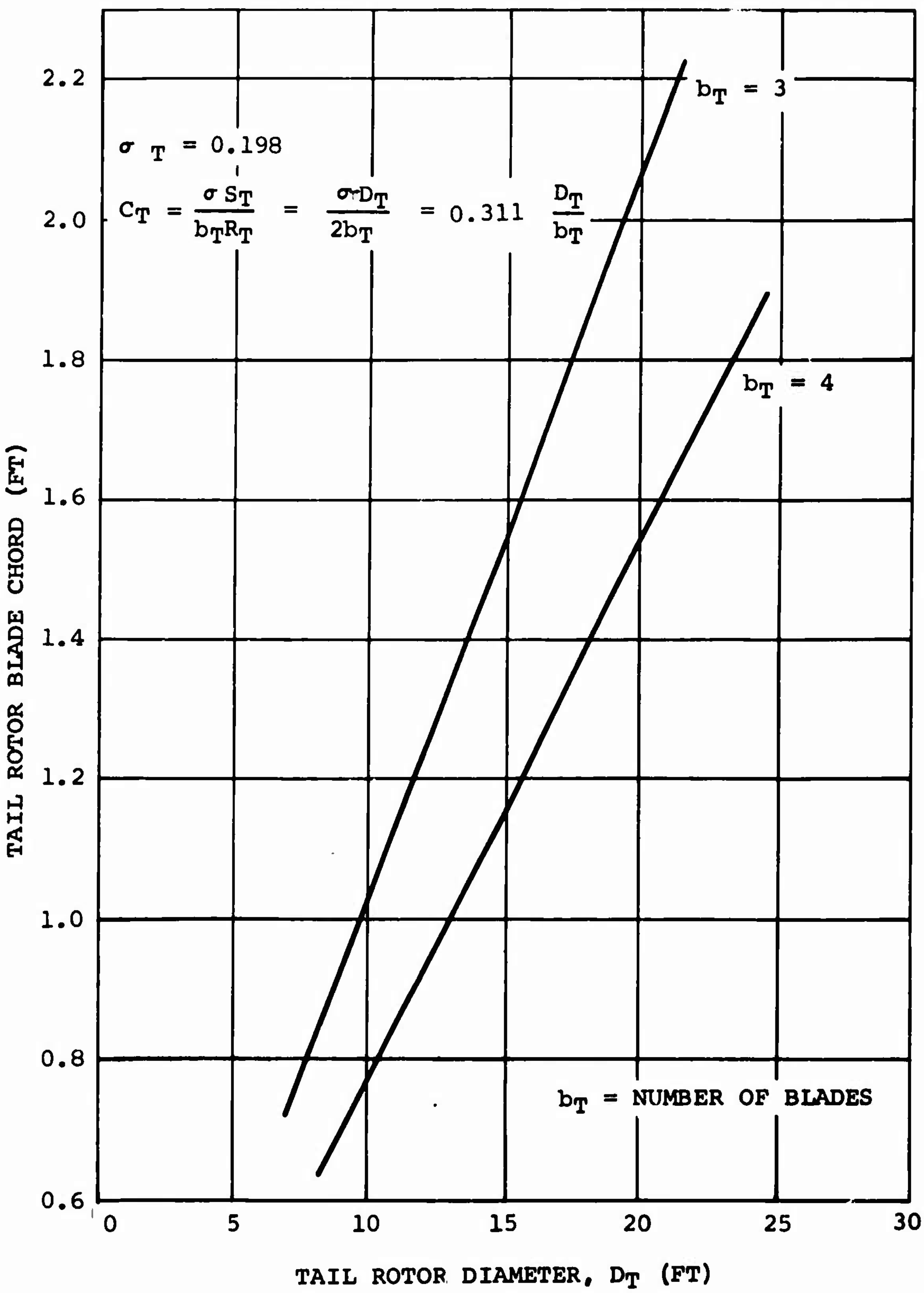


Figure 44. Tail Rotor Blade Chord vs. Tail Rotor Diameter

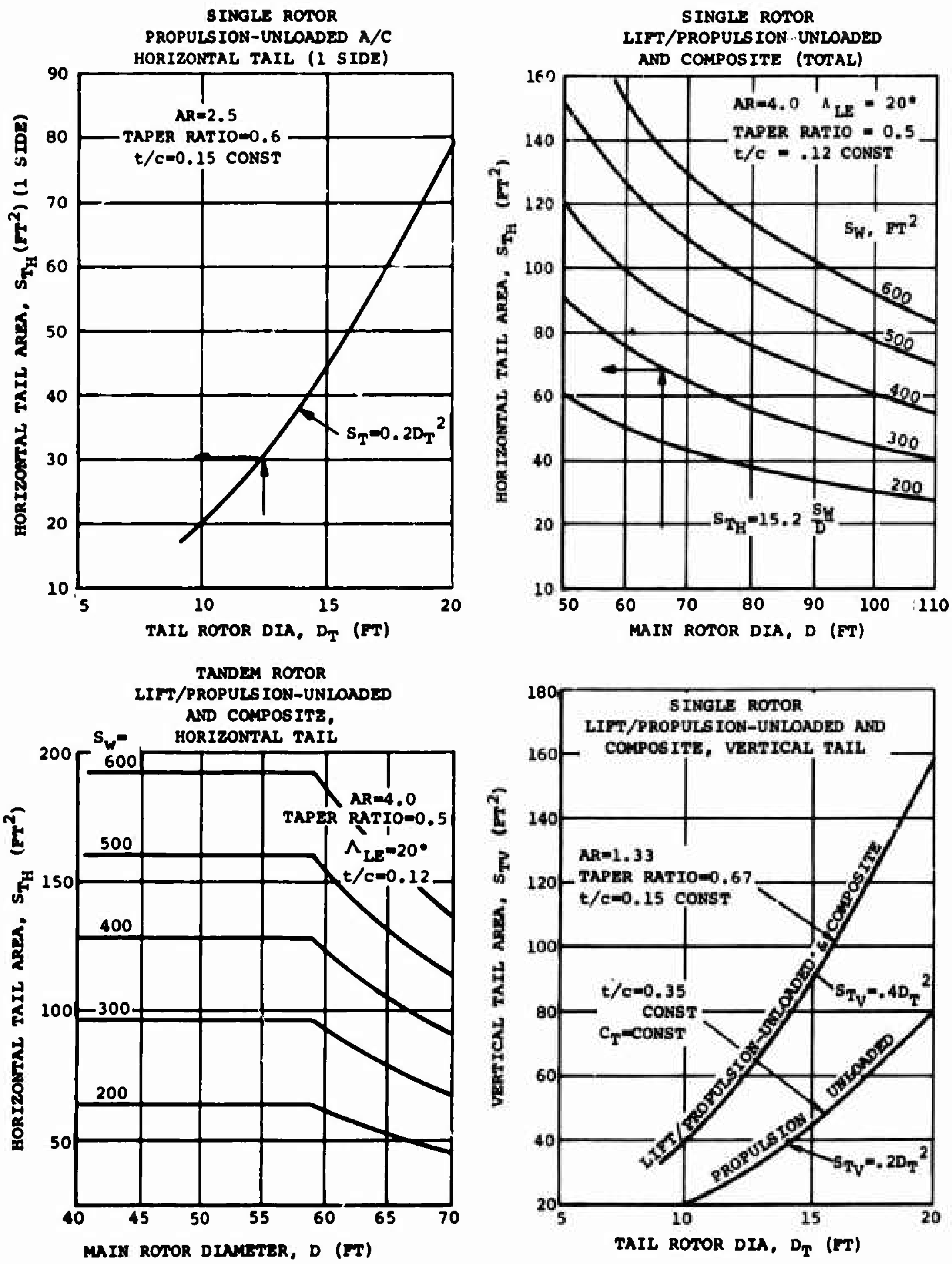


Figure 45. Tail Surface Data

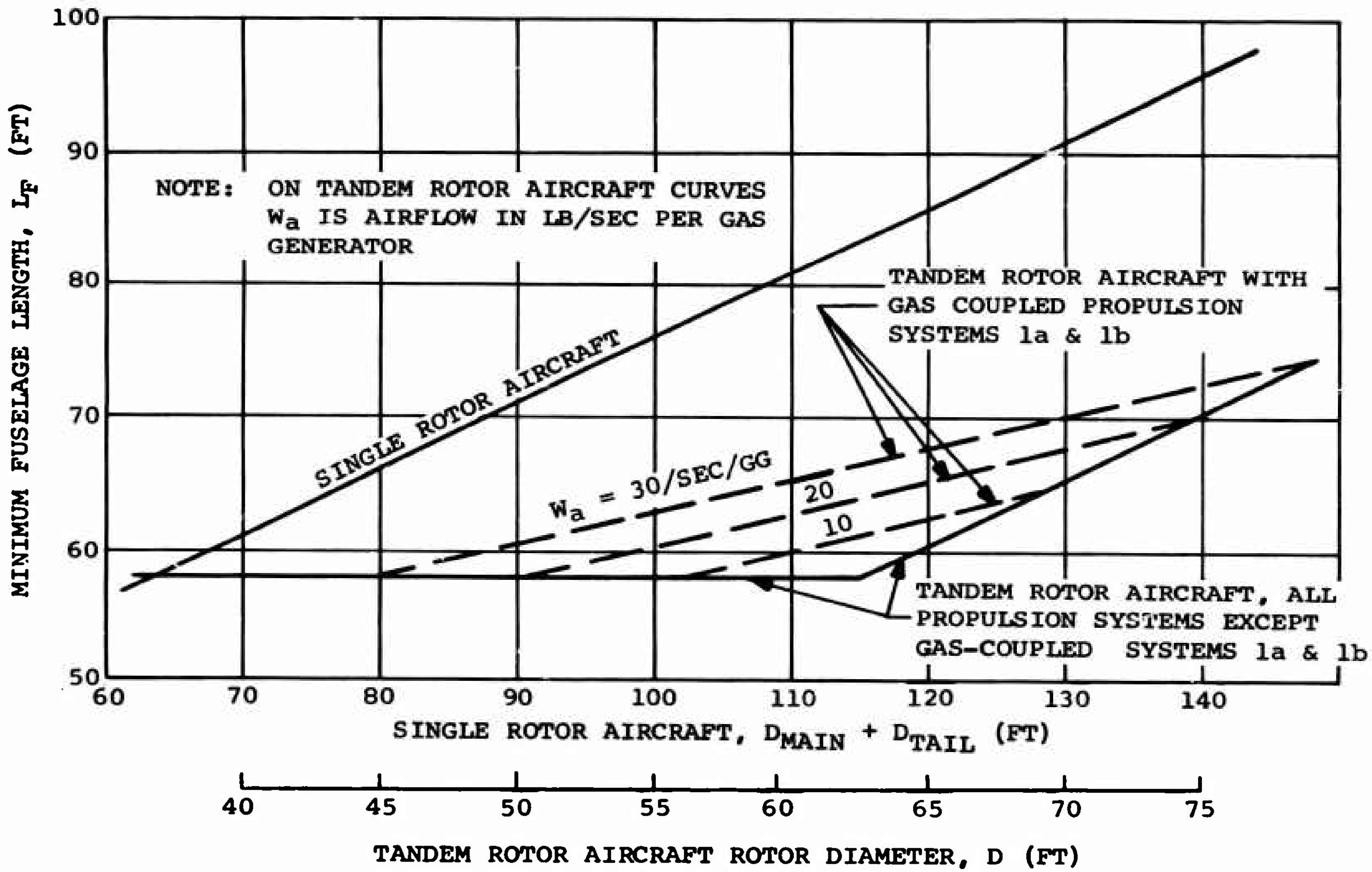


Figure 46 . Minimum Fuselage Length Required vs. Rotor Size, Single and Tandem Rotor Aircraft

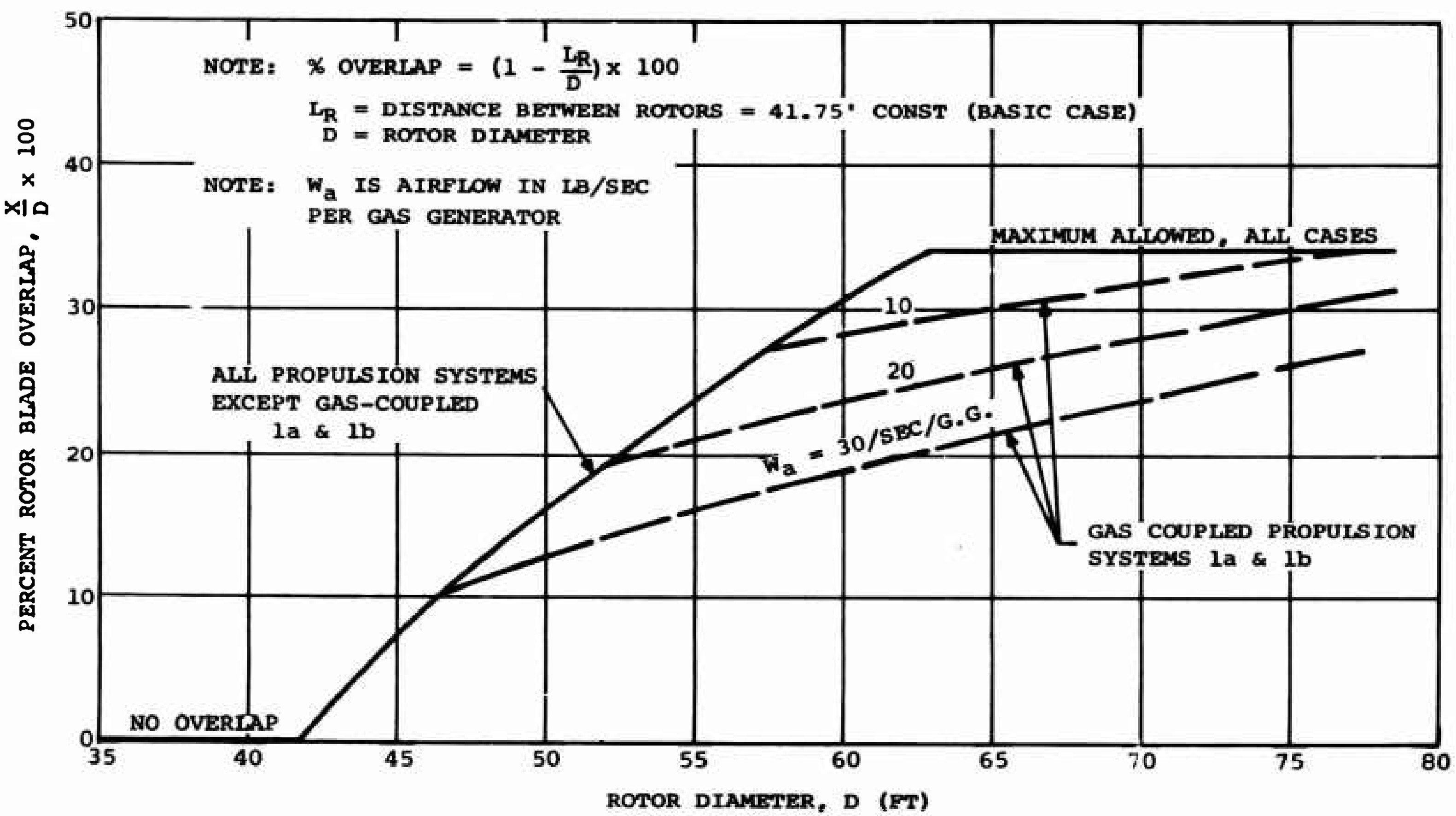


Figure 47 . Percent Rotor Blade Overlap vs. Rotor Diameter, Tandem Rotor Aircraft

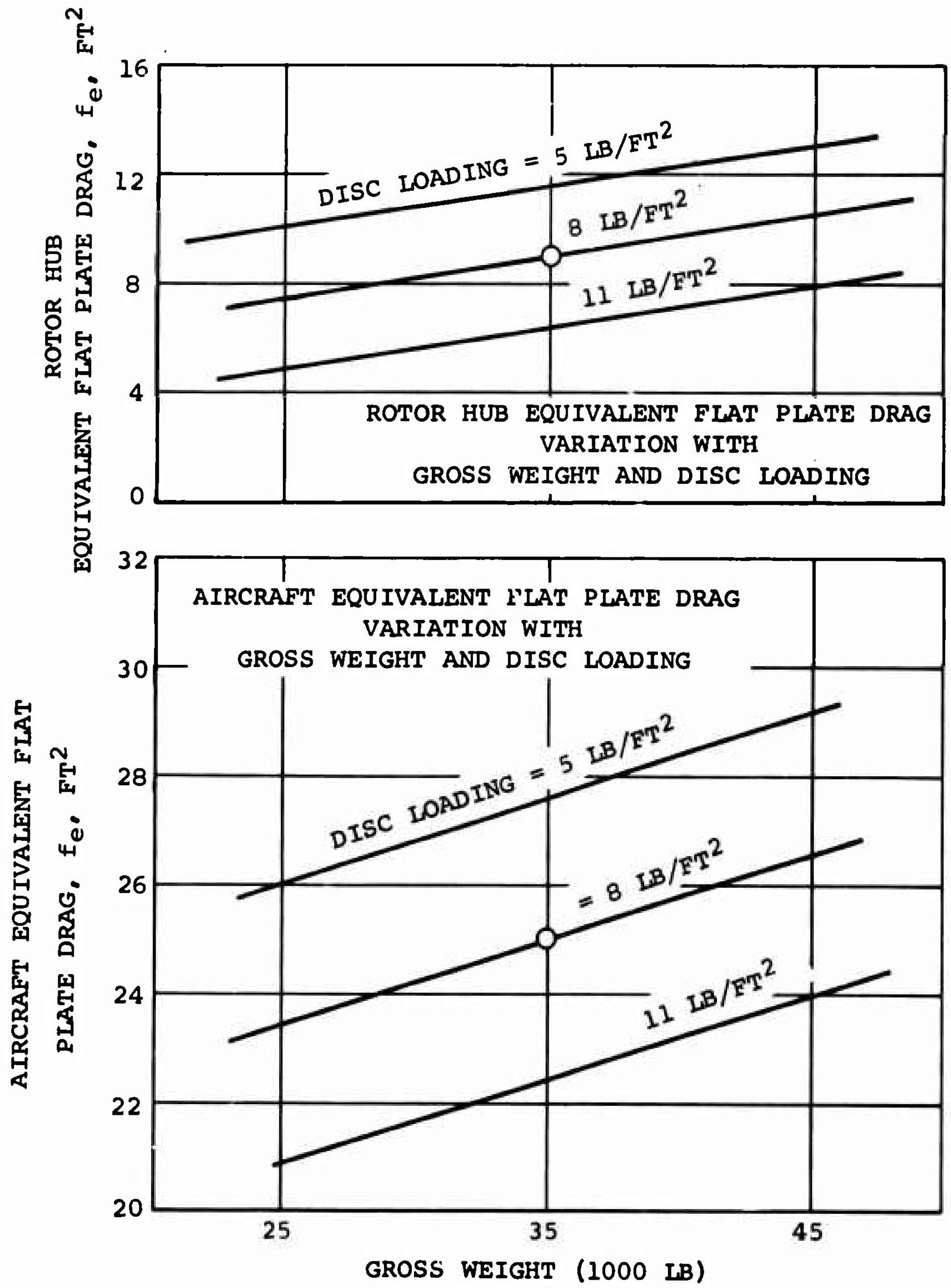


Figure 48. Tandem Rotor Propulsion - Unloaded Compound Aircraft Drag Variations

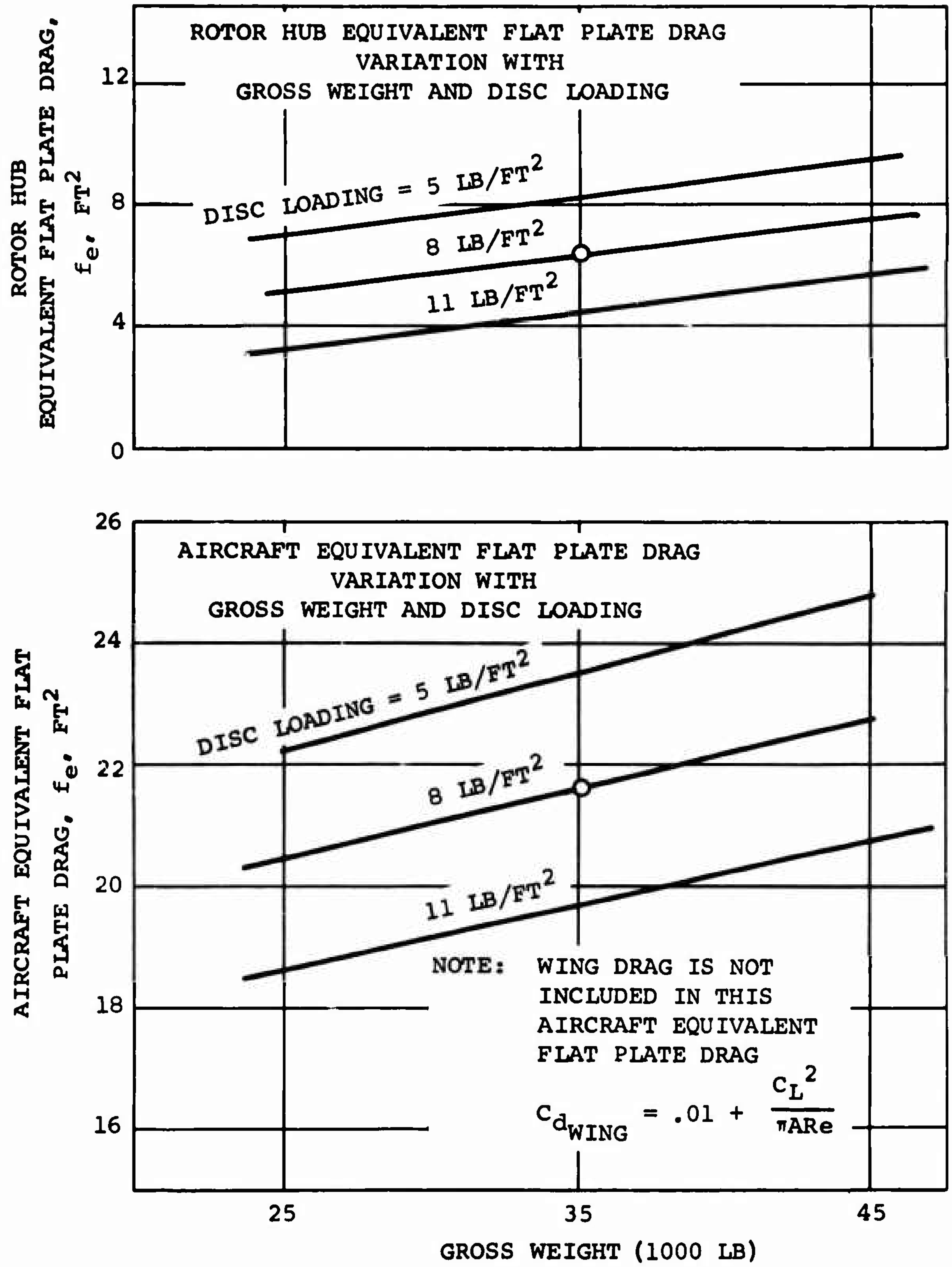


Figure 49. Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Drag Variations

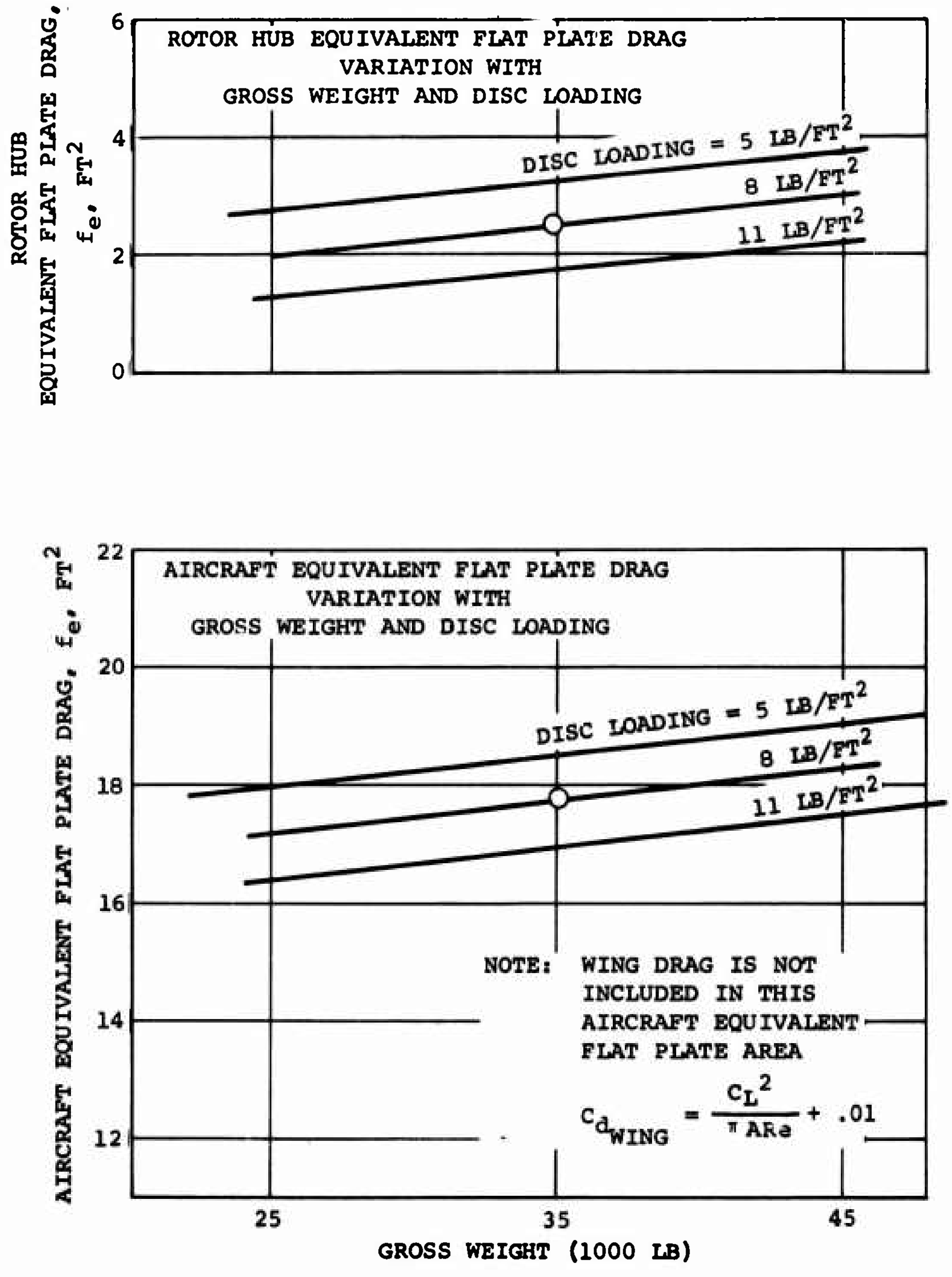


Figure 50. Tandem Rotor Composite Aircraft Drag Variations

Tandem Rotor Propulsion Unloaded Compound Aircraft - Propulsion System 2a

Item		Sample Calculation
1	Design flight condition	Hover, 6000', 95°F
2	Type of aircraft	Propulsion-unloaded compound
3	Disc loading (lb/ft^2)	8
4	Gross weight GW (lb)	25,000
5	Rotor diameter, D (ft)	44.5
6	Rotor radius, R (ft)	22.25
7	Wing loading, GW/S_W (psf) (where applicable)	-
8	Wing area, S_W (ft^2) (where applicable)	-
9	Download (percent)	6.45
10	Number rotor blades/aircraft, b_t	6
11	Parameter GW/b_tD	93.7
12	Rotor blade chord, c (ft)	1.81
13	Blade loading, $W_b = GW/b_t\ cr$ (lb/ft^2)	104
14	Overlap (percent)	12.6
15	Rotor power loading (lb/rotor HP)	7.08
16	Rotor HP required (SHP) at 6000 ft, 95°F	3535
17	Transmission efficiency, η_T (rotor to cross-shaft input)	0.95
18	Total cross-shaft power required (SHP) 6000 ft, 95°F	3720
19	Power lapse factor, 6000 ft, 95°F to SL Std	0.78
20	Total cross-shaft power required, SL Std (SHP)	4775
21	Fan bypass ratio	(not required)
22	Total gas generator airflow required (lb/sec)	28.0
23	Number of gas generators	2
24	Uncorrected airflow required per gas generator W_a' (lb/sec)	14.0
25	Gas generator airflow correction for 60 AHP (lb/sec)	0.15
26	Corrected airflow required per gas generator SL Std W_a' (lb/sec)	14.15

Figure 51. Sample Gas Generator Sizing Data Sheet

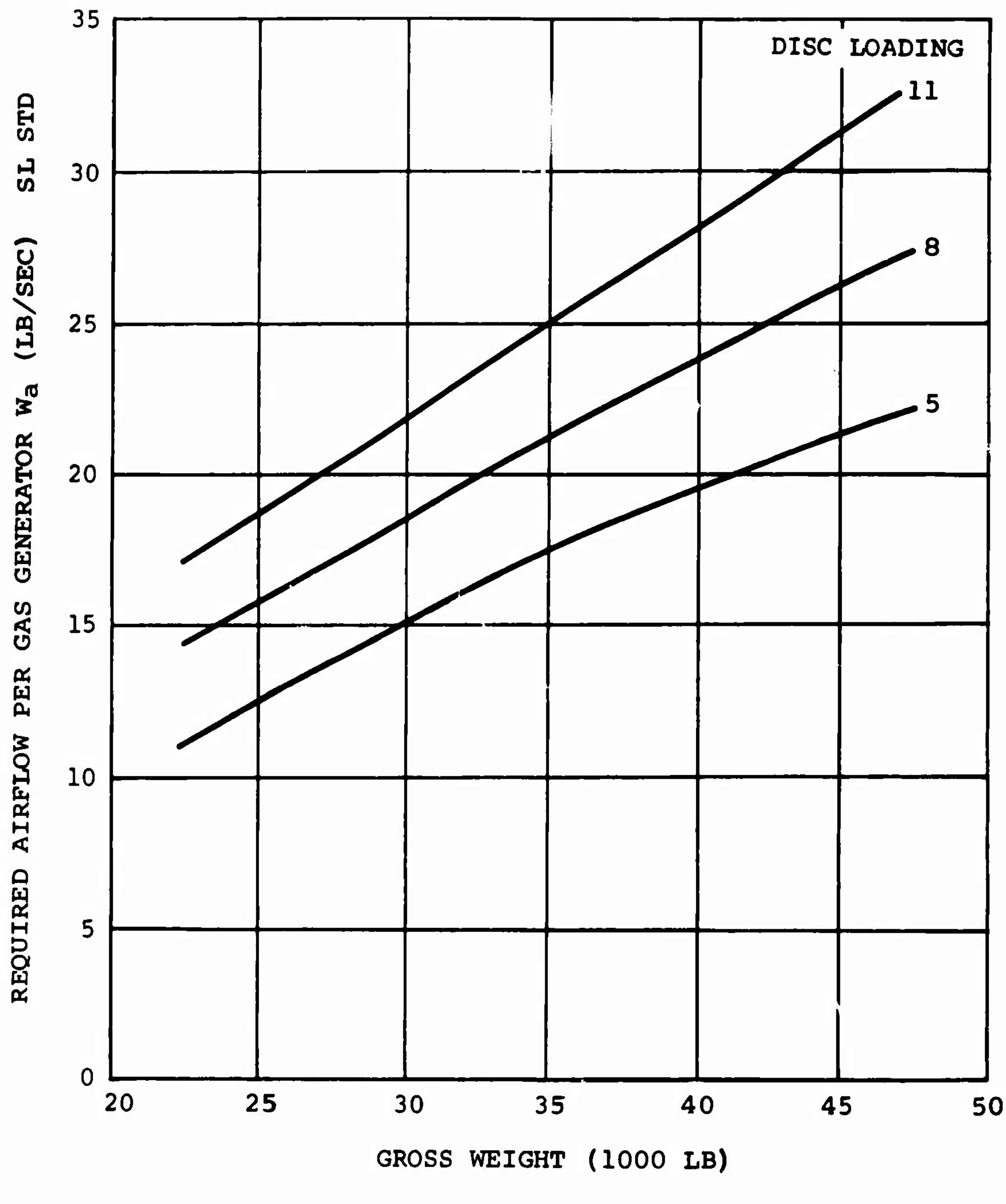


Figure 52. Required Sea Level Standard Gas Generator Airflow to Meet the 6000-Foot, 95°F Hover Requirement - Single Rotor Propulsion-Unloaded Compound Aircraft Propulsion Systems 1a and 1b (Integrated Gas-Coupled)

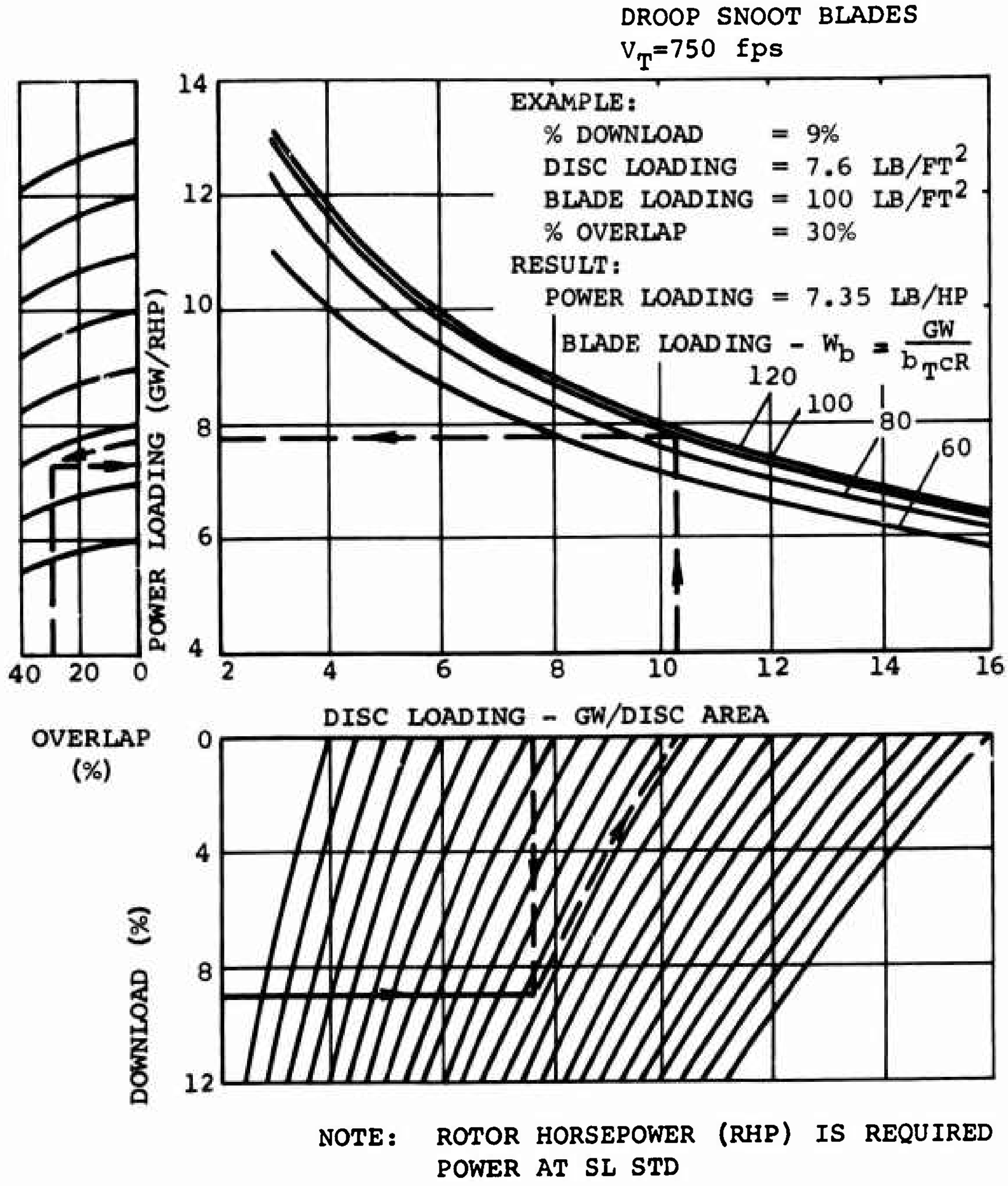
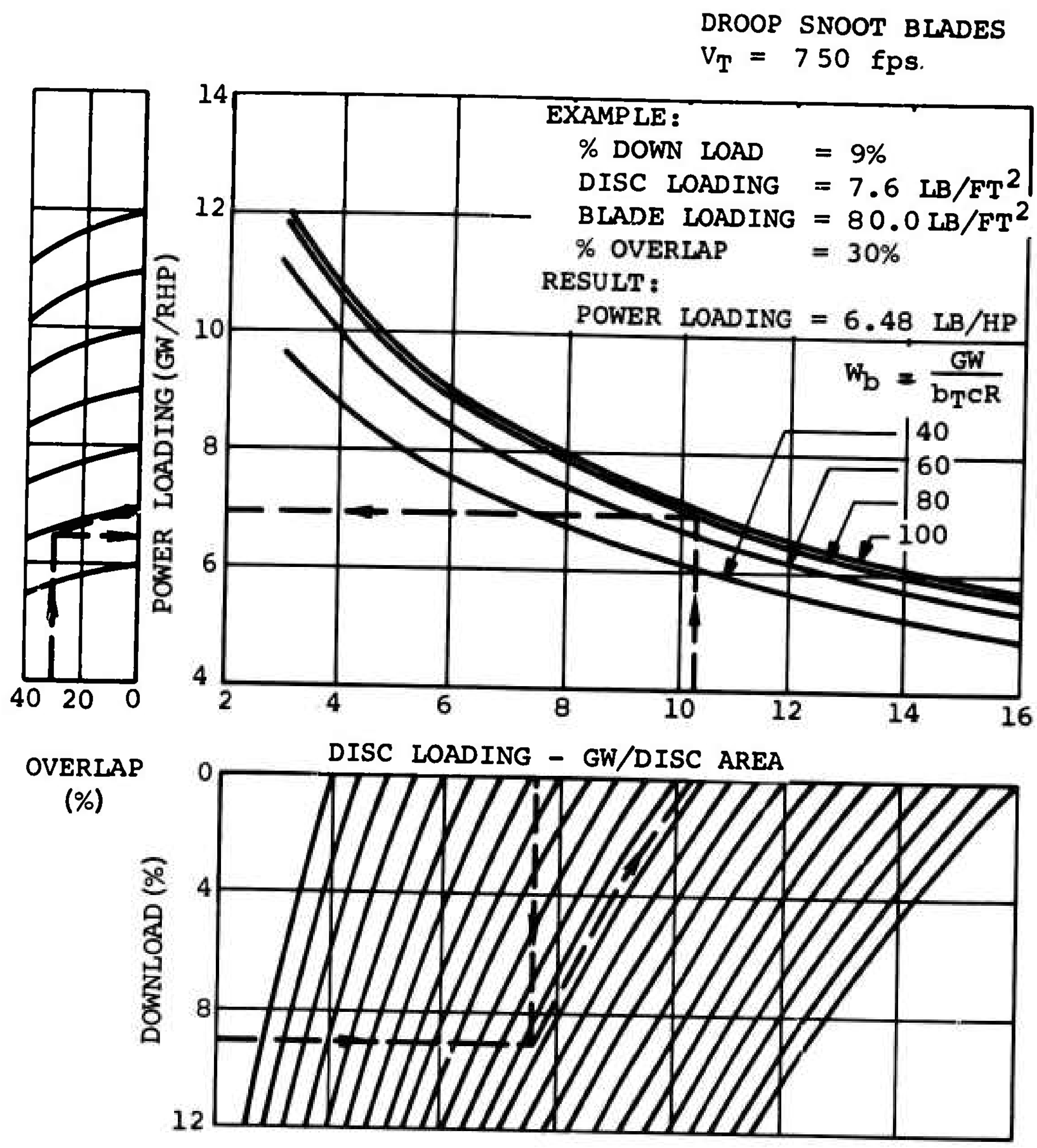


Figure 53. Determination of Required Rotor Power Loading at Sea Level Standard Conditions



NOTE: ROTOR HORSEPOWER (RHP) IS REQUIRED POWER AT 6000 FEET, 95°F

Figure 54. Determination of Required Rotor Power Loading at 6000-Foot, 95°F Conditions

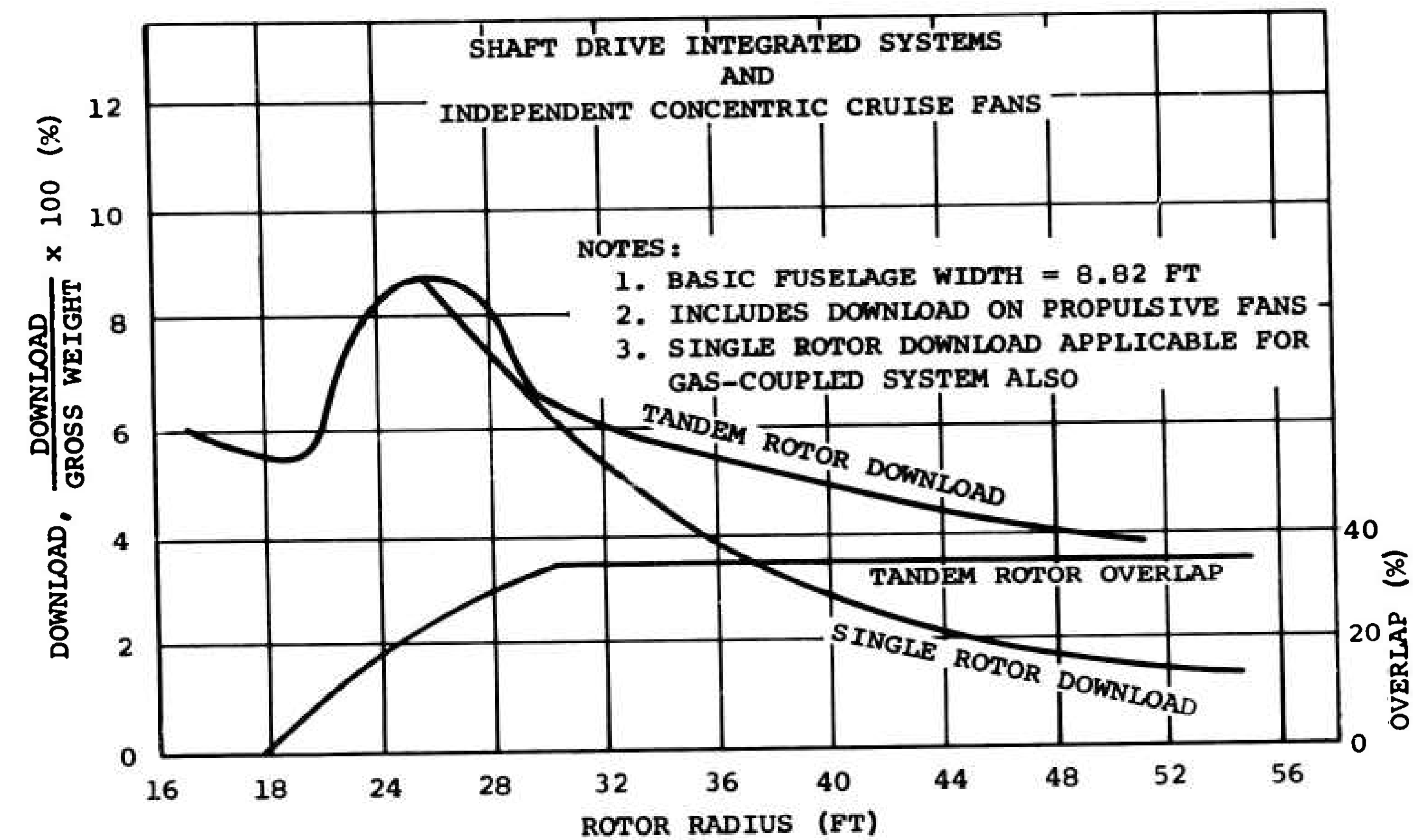


Figure 55. Effect of Rotor Radius on Percent Download for Propulsion-Unloaded Configurations

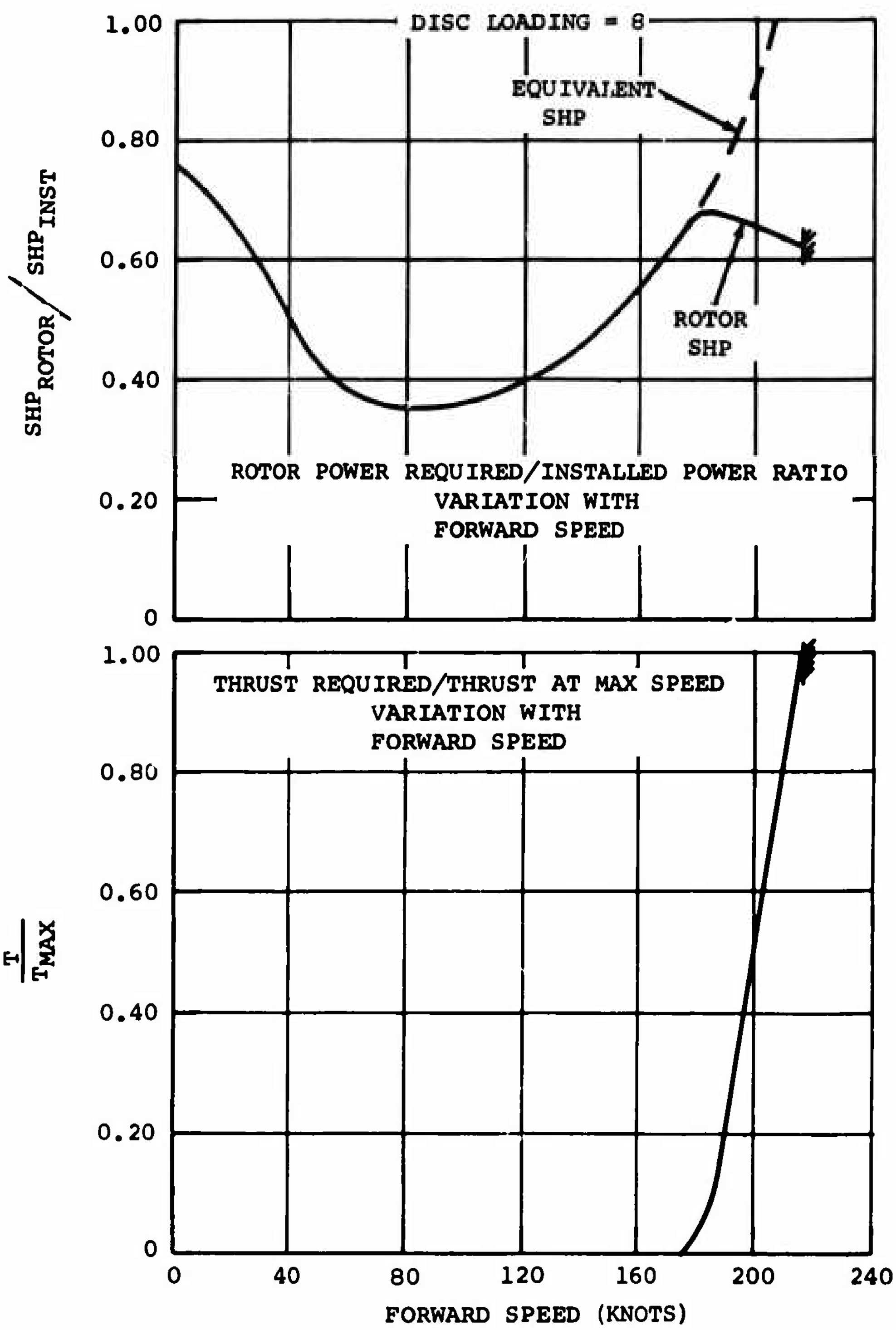


Figure 56. Power Requirements of Propulsion-Unloaded Aircraft

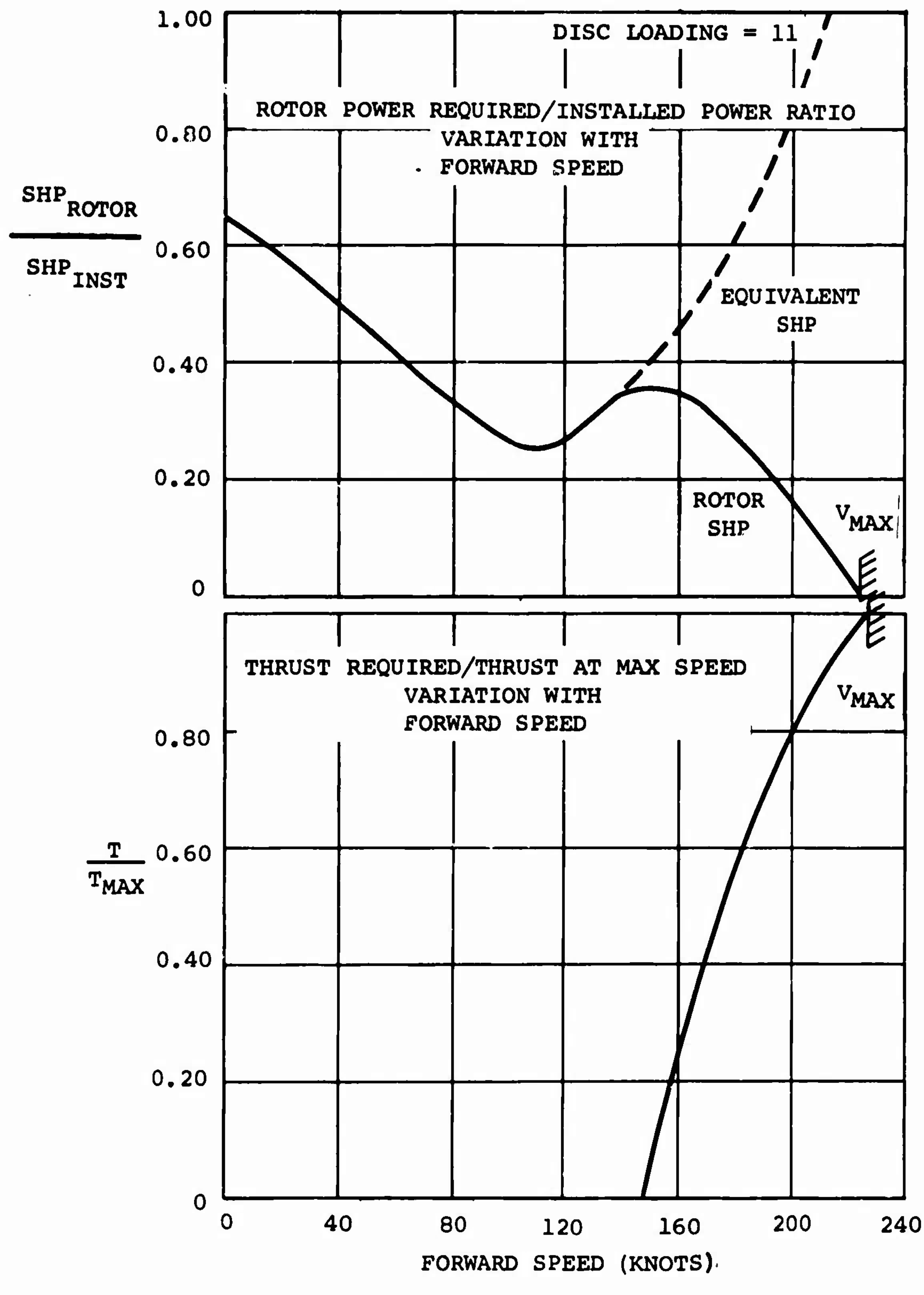


Figure 57. Power Requirements of Lift/Propulsion-Unloaded Aircraft

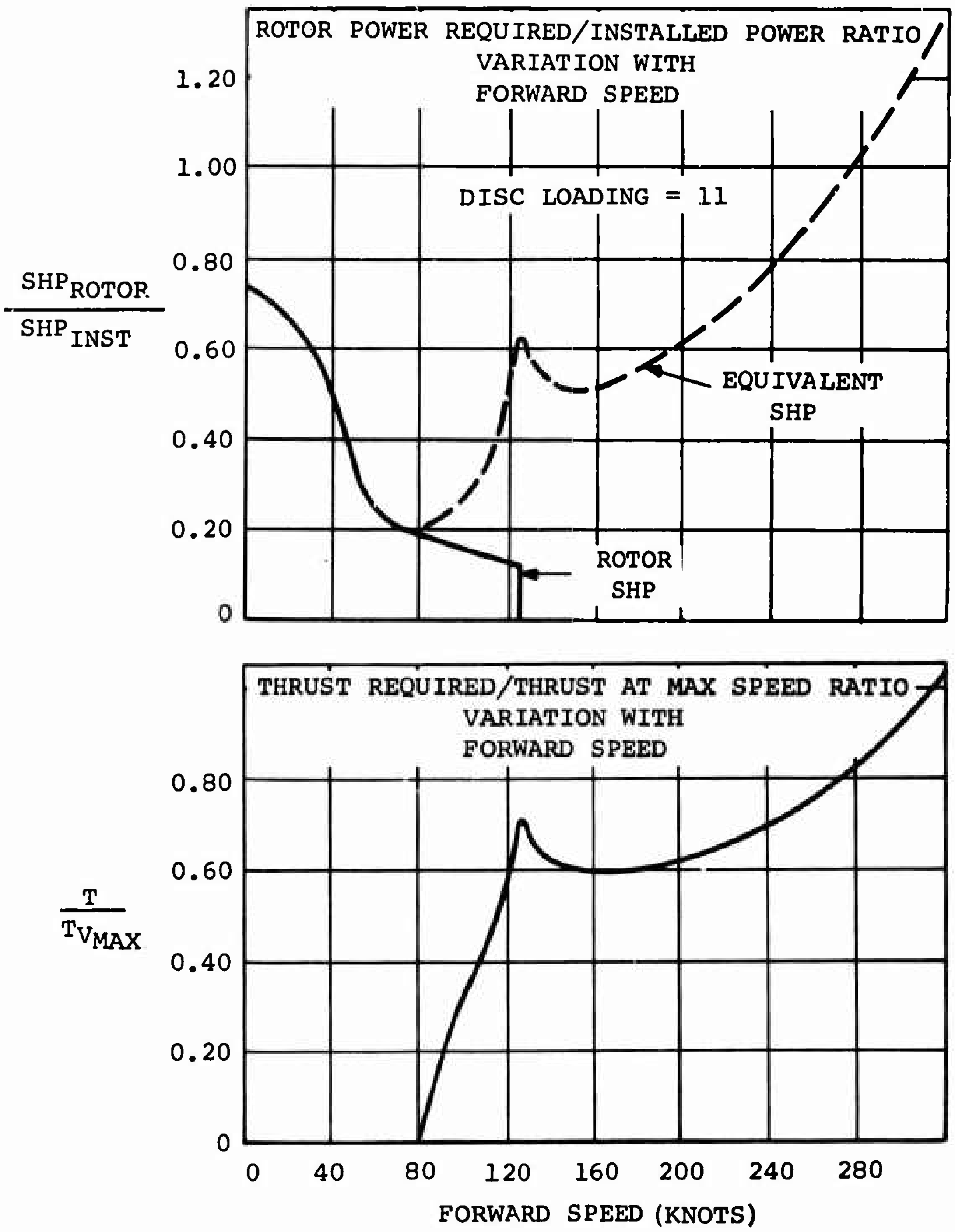
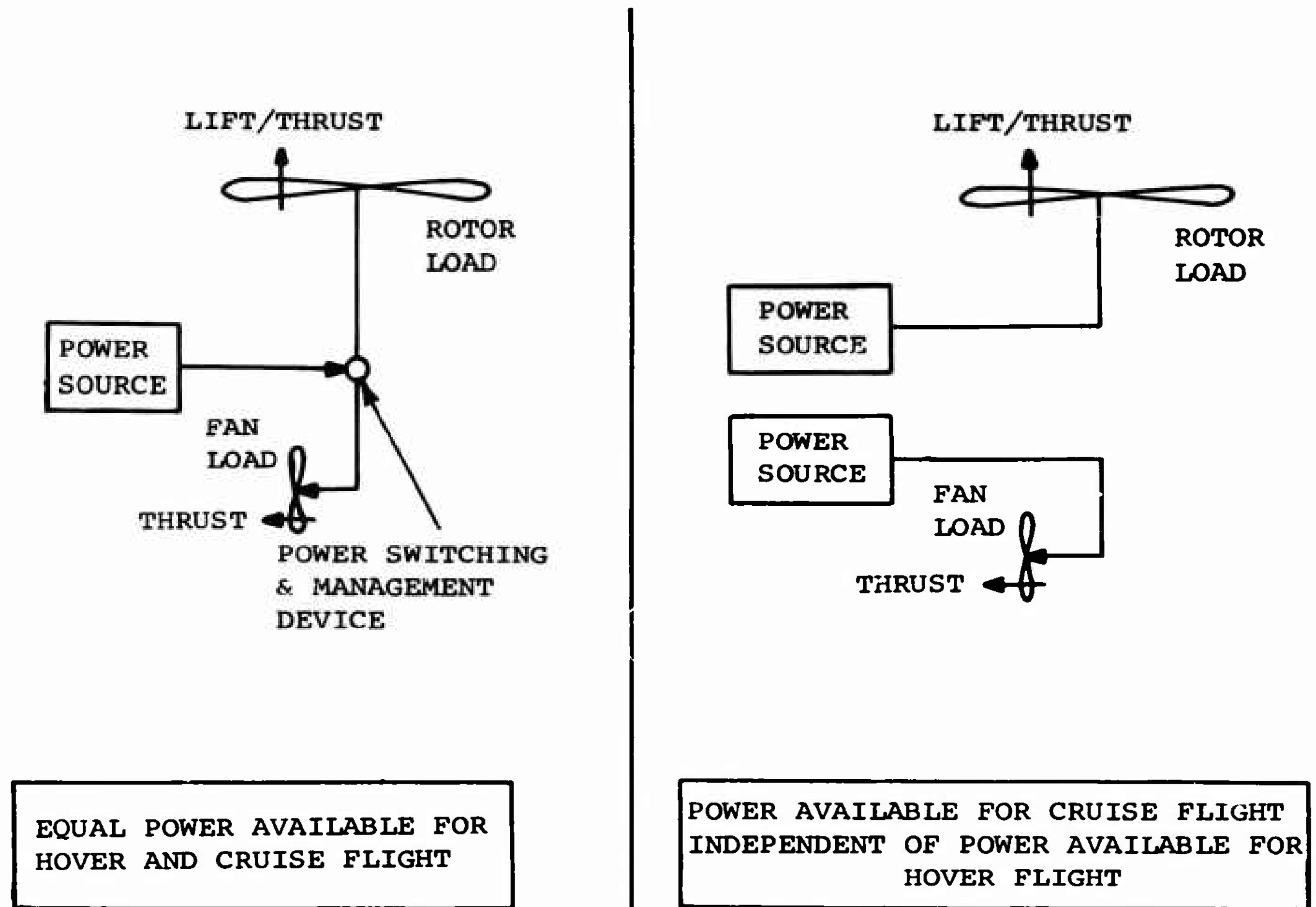


Figure 58. Power Requirements for Composite Aircraft



Integrated Propulsion System

Independent Propulsion System

Figure 59. Fundamental Systems Concepts

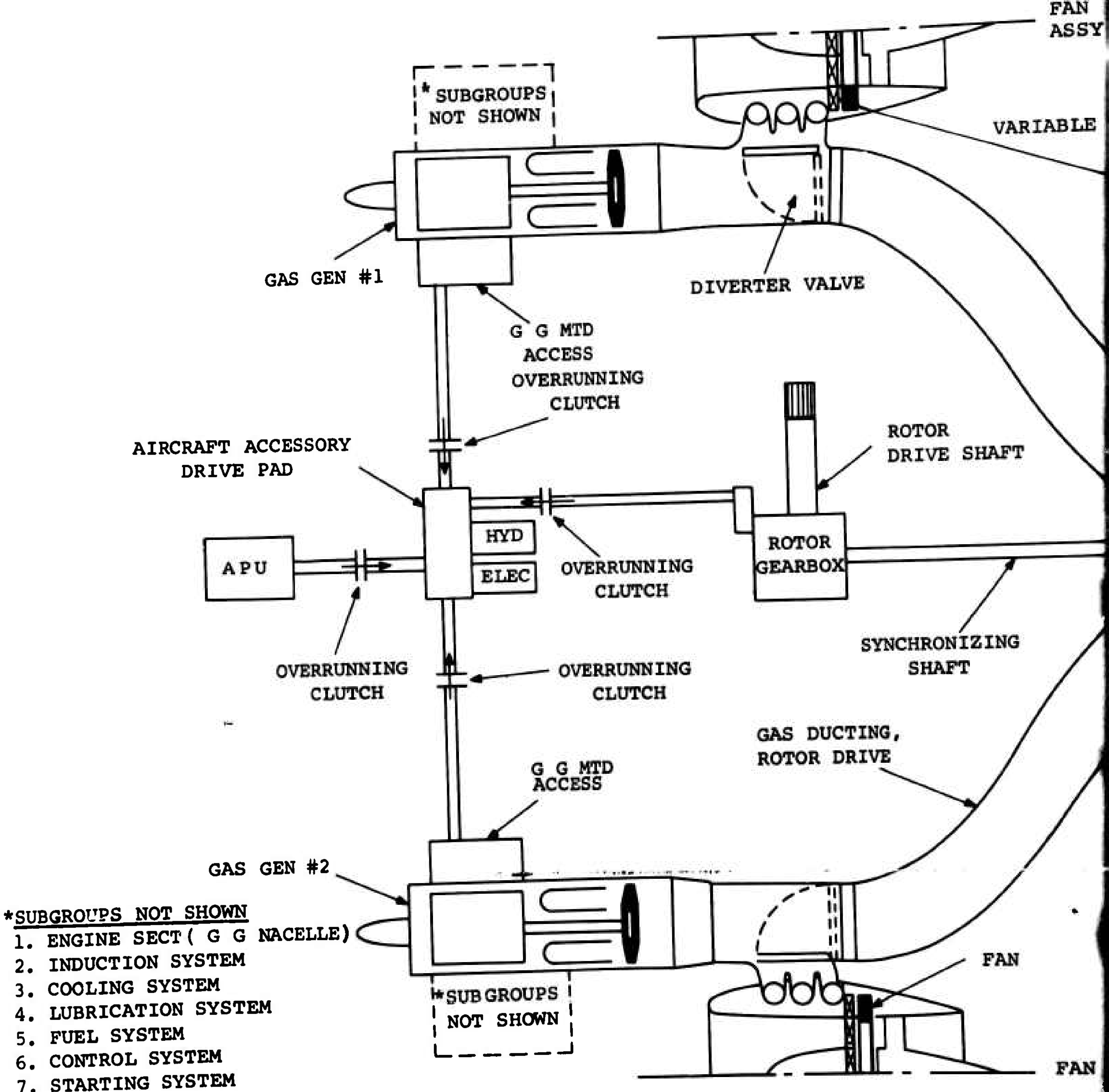
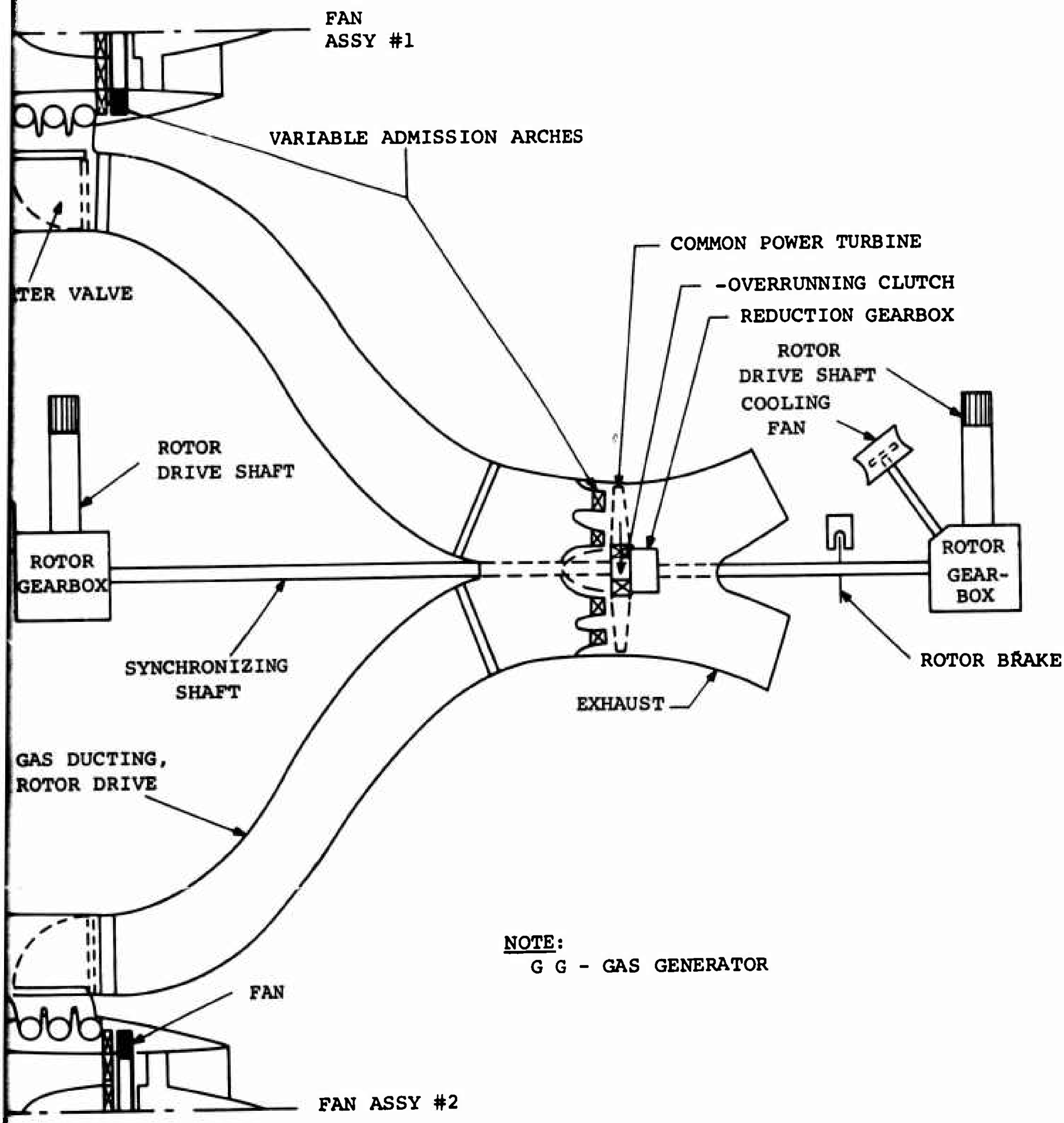


Figure 60. Integrated Lift/Propulsion System Gas-Coupled Remote Tip-Driven Cruise Fans - Tandem Rotor Aircraft (System 1a), Compound and Composite Types

A



FAN
ASSY #1
VARIABLE ADMISSION ARCHES
TER VALVE
COMMON POWER TURBINE
-OVERRUNNING CLUTCH
REDUCTION GEARBOX
ROTOR
DRIVE SHAFT
COOLING
FAN
ROTOR
DRIVE SHAFT
ROTOR
GEARBOX
ROTOR
GEAR-
BOX
ROTOR BRAKE
SYNCHRONIZING
SHAFT
EXHAUST
GAS DUCTING,
ROTOR DRIVE
NOTE:
G G - GAS GENERATOR
FAN
FAN ASSY #2

d

B

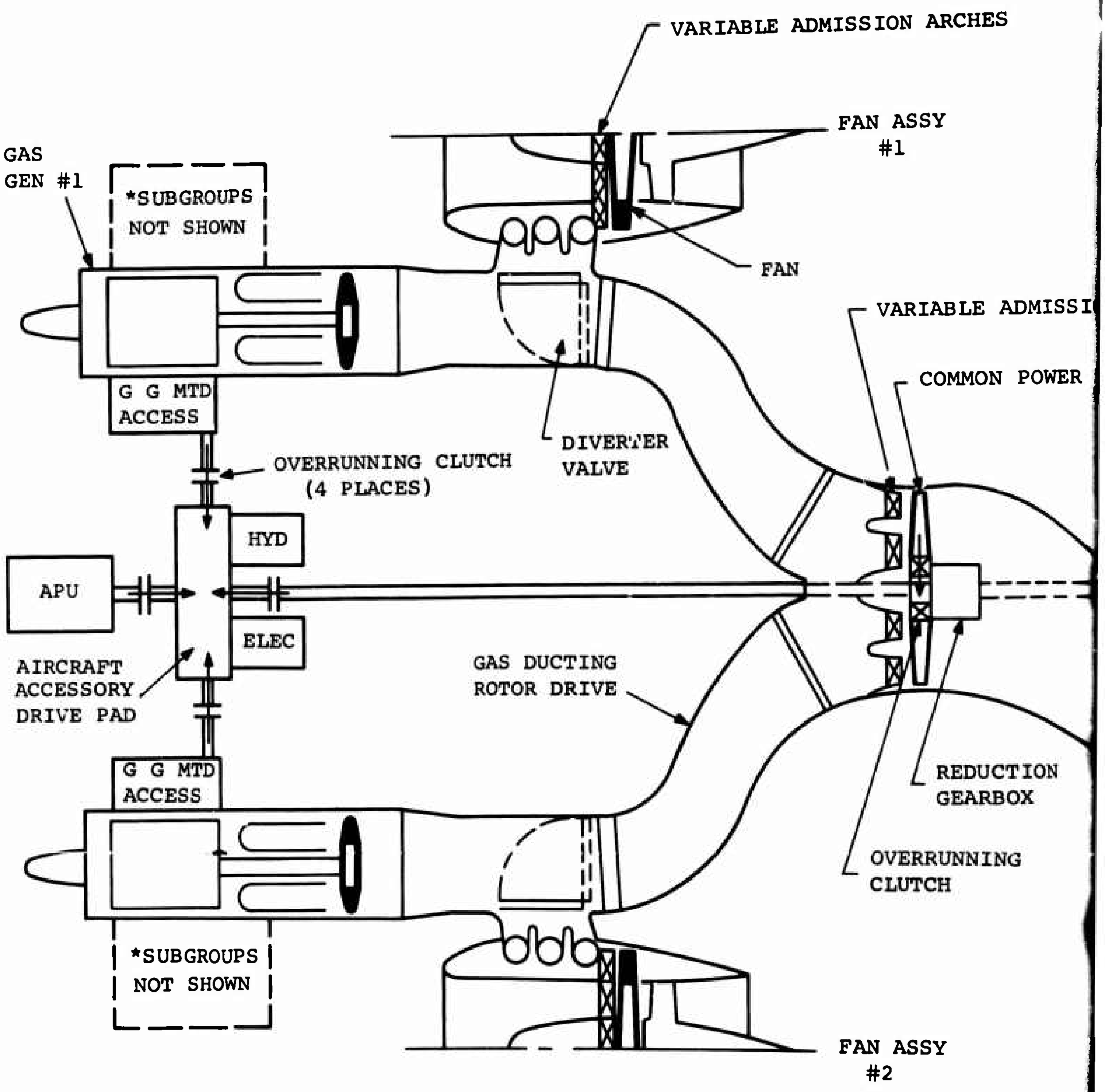
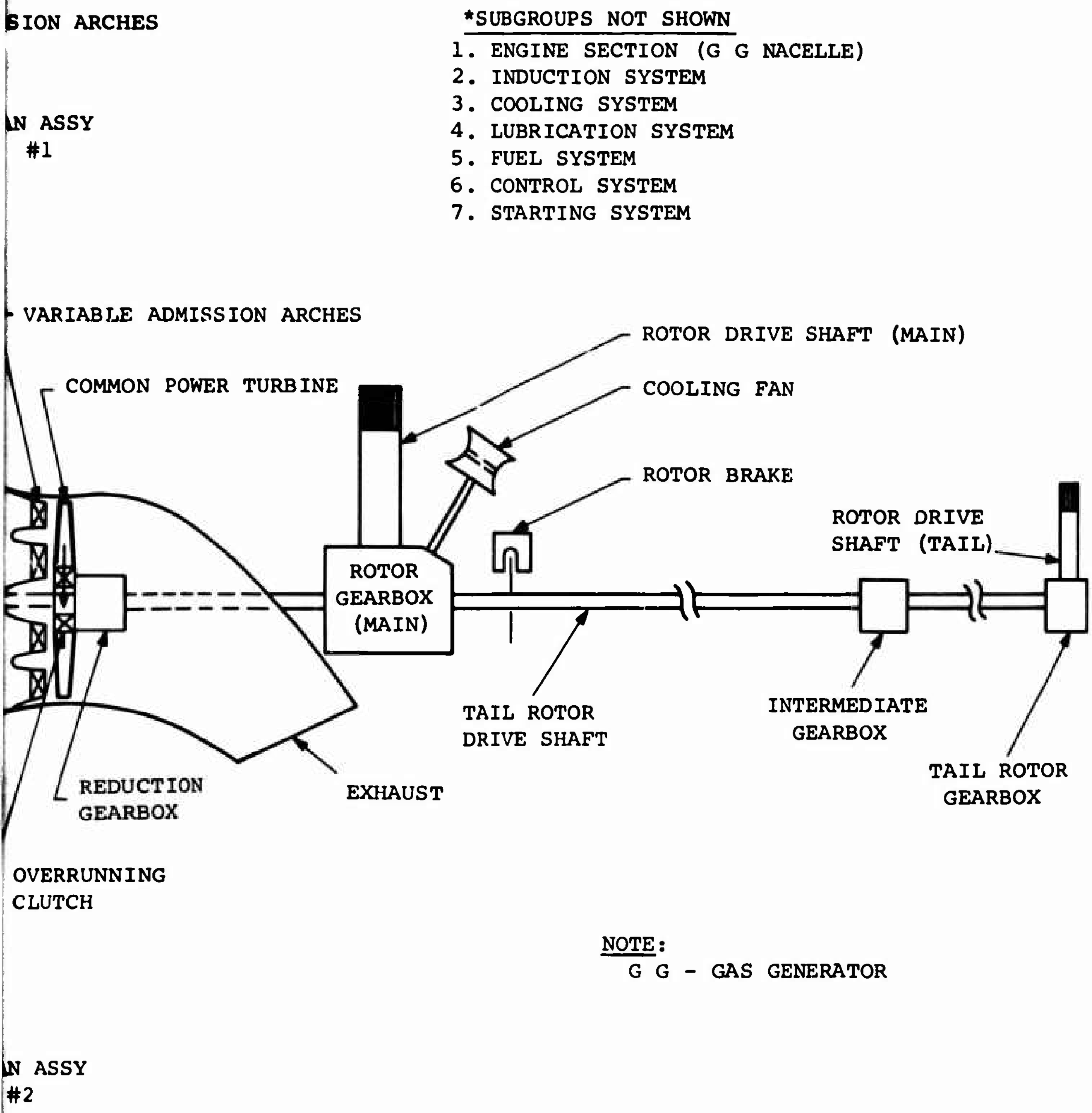


Figure 61. Integrated Lift/Propulsion System Gas-Coupled Remote Tip-Driven Cruise Fans - Single Rotor Aircraft (System 1a), Compound and Composite Types

A

*SUBGROUPS NOT SHOWN

1. ENGINE SECTION (G G NACELLE)
2. INDUCTION SYSTEM
3. COOLING SYSTEM
4. LUBRICATION SYSTEM
5. FUEL SYSTEM
6. CONTROL SYSTEM
7. STARTING SYSTEM



NOTE:
G G - GAS GENERATOR

3

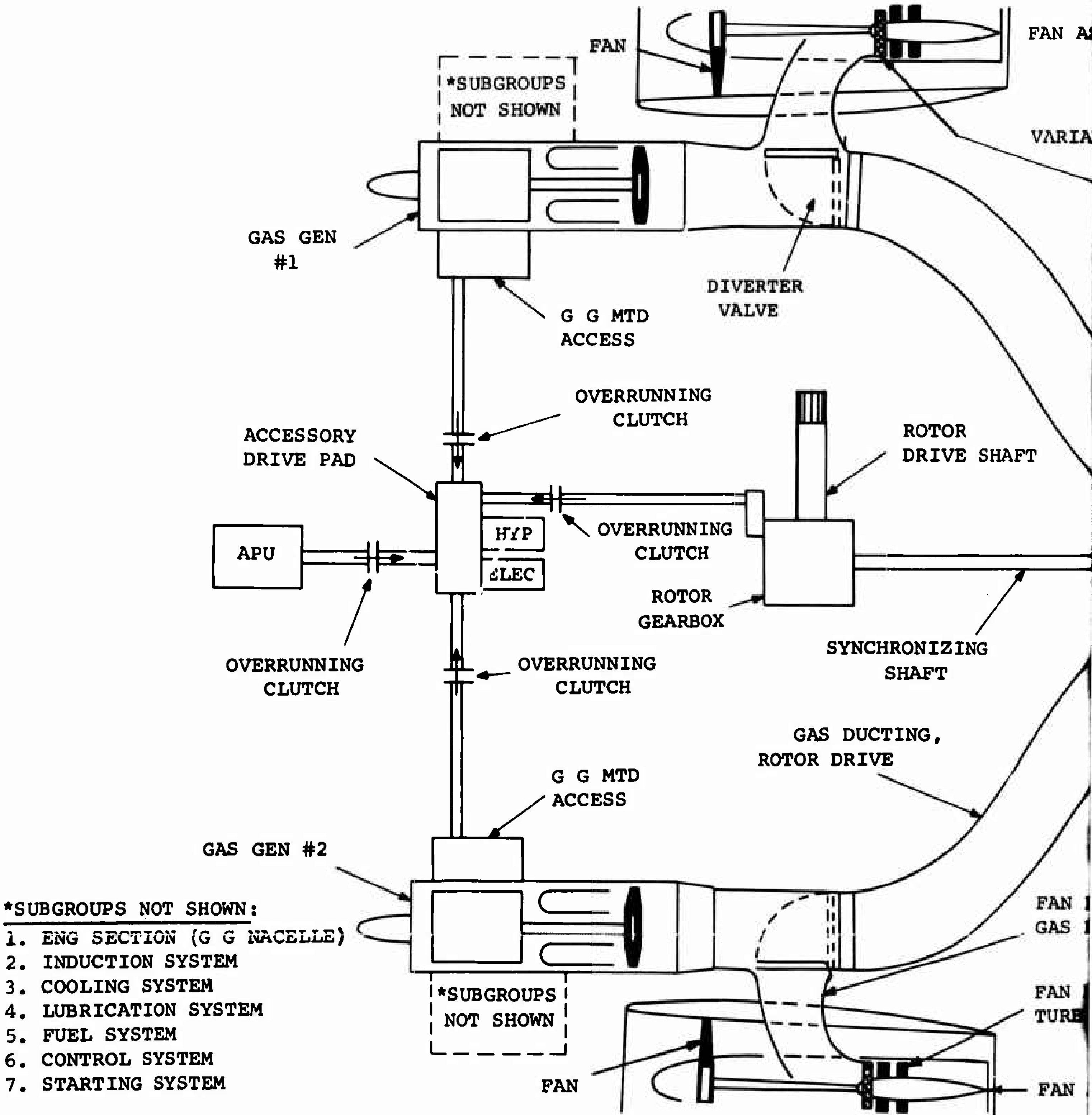
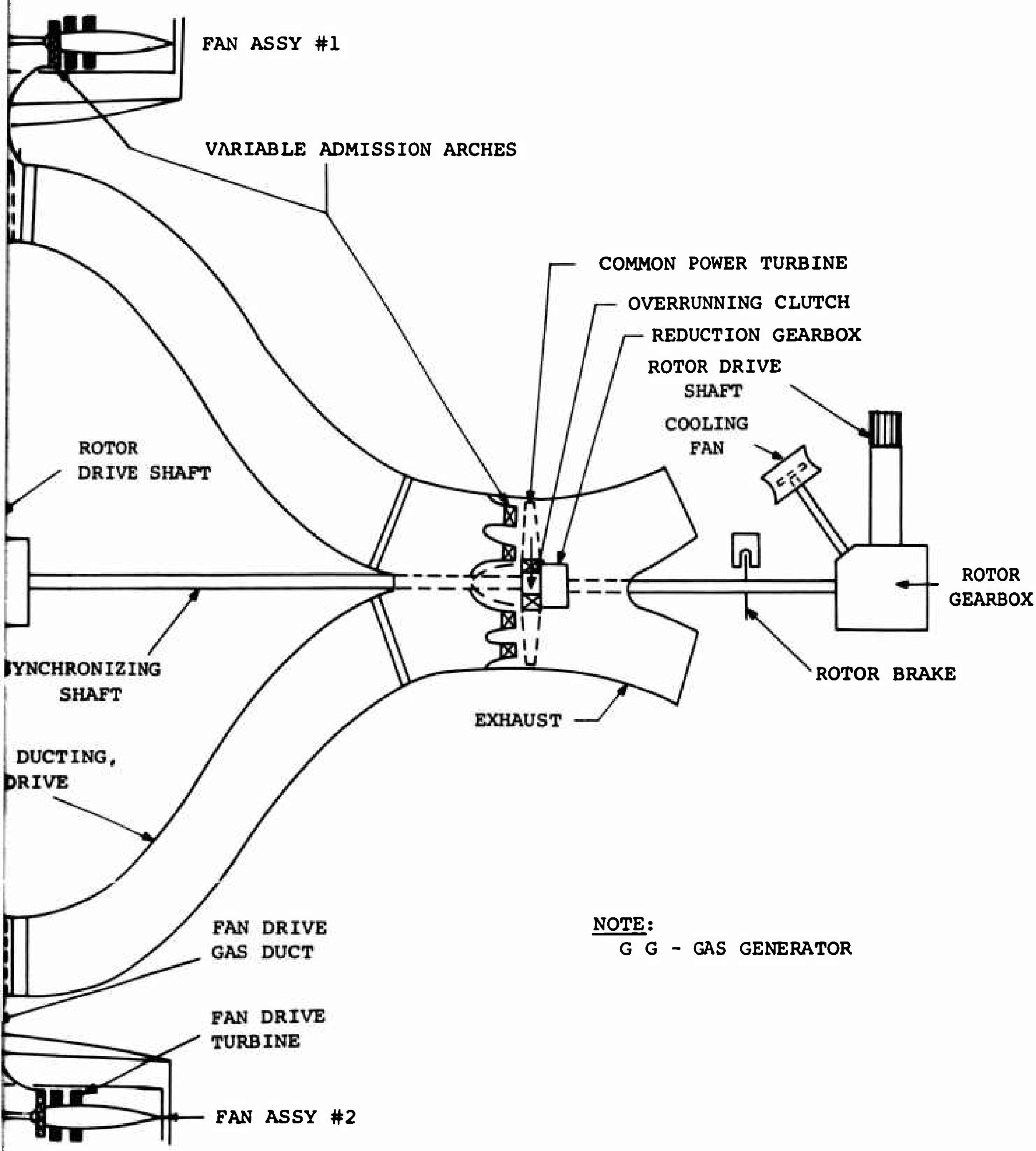
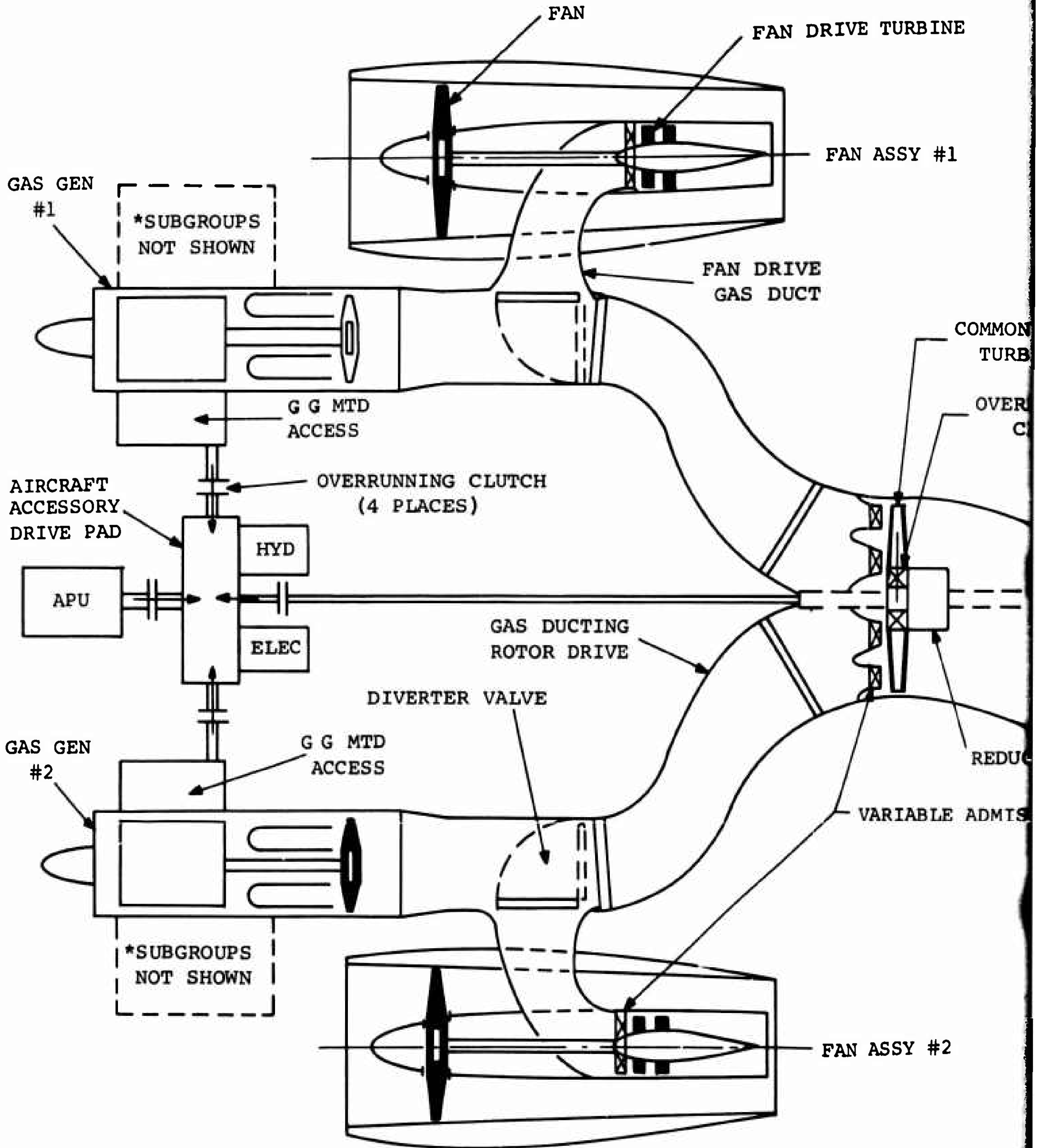


Figure 62. Integrated Lift/Propulsion System Gas-Coupled Remote Axially Driven Cruise Fan (System 1b) - Tandem Rotor Aircraft, Compound and Composite Types

A


FAN ASSY #1
VARIABLE ADMISSION ARCHES
COMMON POWER TURBINE
OVERRUNNING CLUTCH
REDUCTION GEARBOX
ROTOR DRIVE SHAFT
COOLING FAN
ROTOR DRIVE SHAFT
ROTOR GEARBOX
SYNCHRONIZING SHAFT
ROTOR BRAKE
EXHAUST
DUCTING, DRIVE
FAN DRIVE GAS DUCT
NOTE:
G G - GAS GENERATOR
FAN DRIVE TURBINE
FAN ASSY #2



FAN
FAN DRIVE TURBINE
FAN ASSY #1
GAS GEN #1
*SUBGROUPS NOT SHOWN
FAN DRIVE GAS DUCT
COMMON TURB
G G MTD ACCESS
OVER C
OVERRUNNING CLUTCH (4 PLACES)
AIRCRAFT ACCESSORY DRIVE PAD
HYD
APU
ELEC
GAS DUCTING ROTOR DRIVE
DIVERTER VALVE
GAS GEN #2
G G MTD ACCESS
REDU
VARIABLE ADMIS
*SUBGROUPS NOT SHOWN
FAN ASSY #2

A

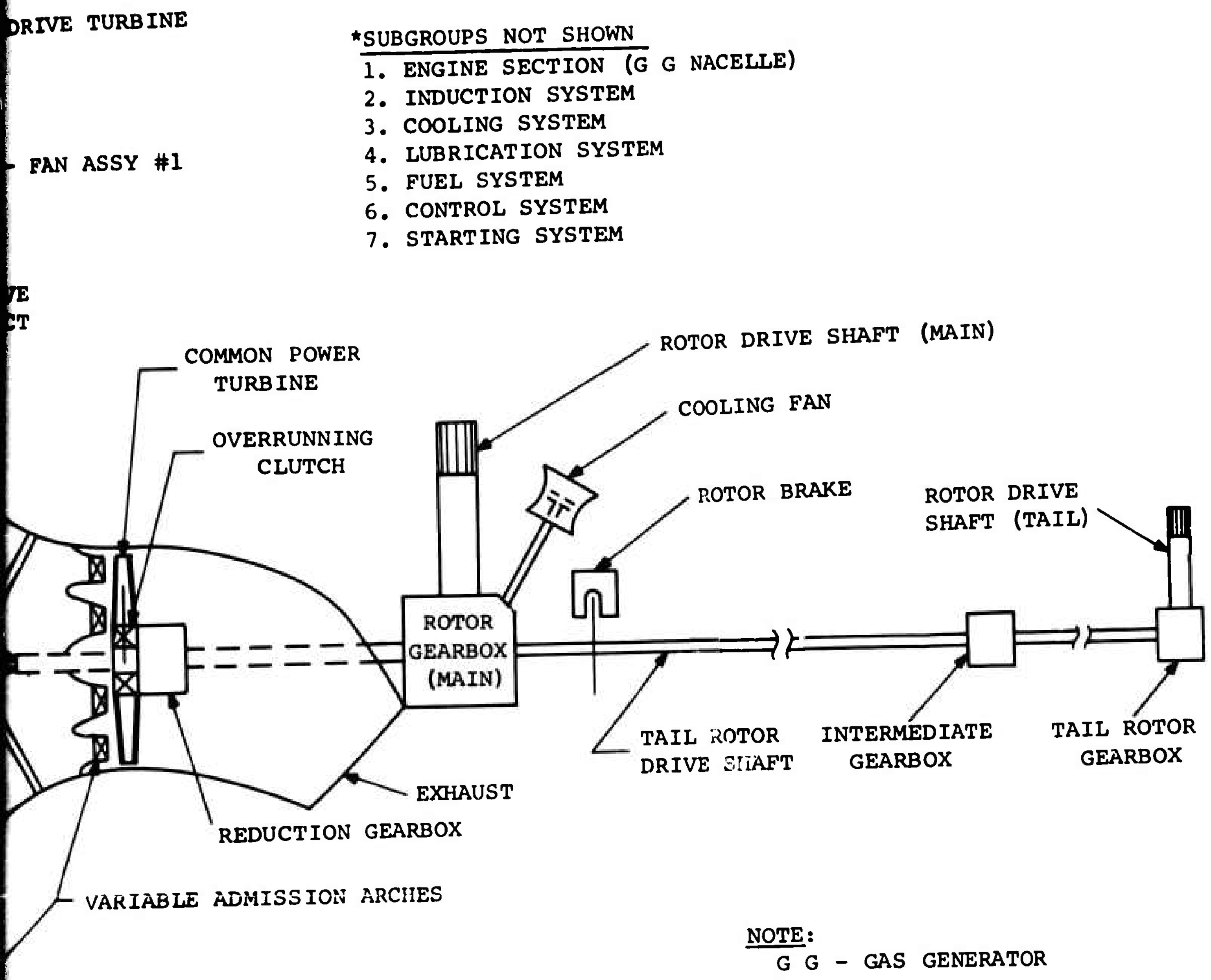


Figure 63. Integrated Lift/Propulsion System Gas-Coupled Remote Axially-Driven Cruise Fans - Single Rotor Aircraft (System 1b), Compound and Composite Types

B

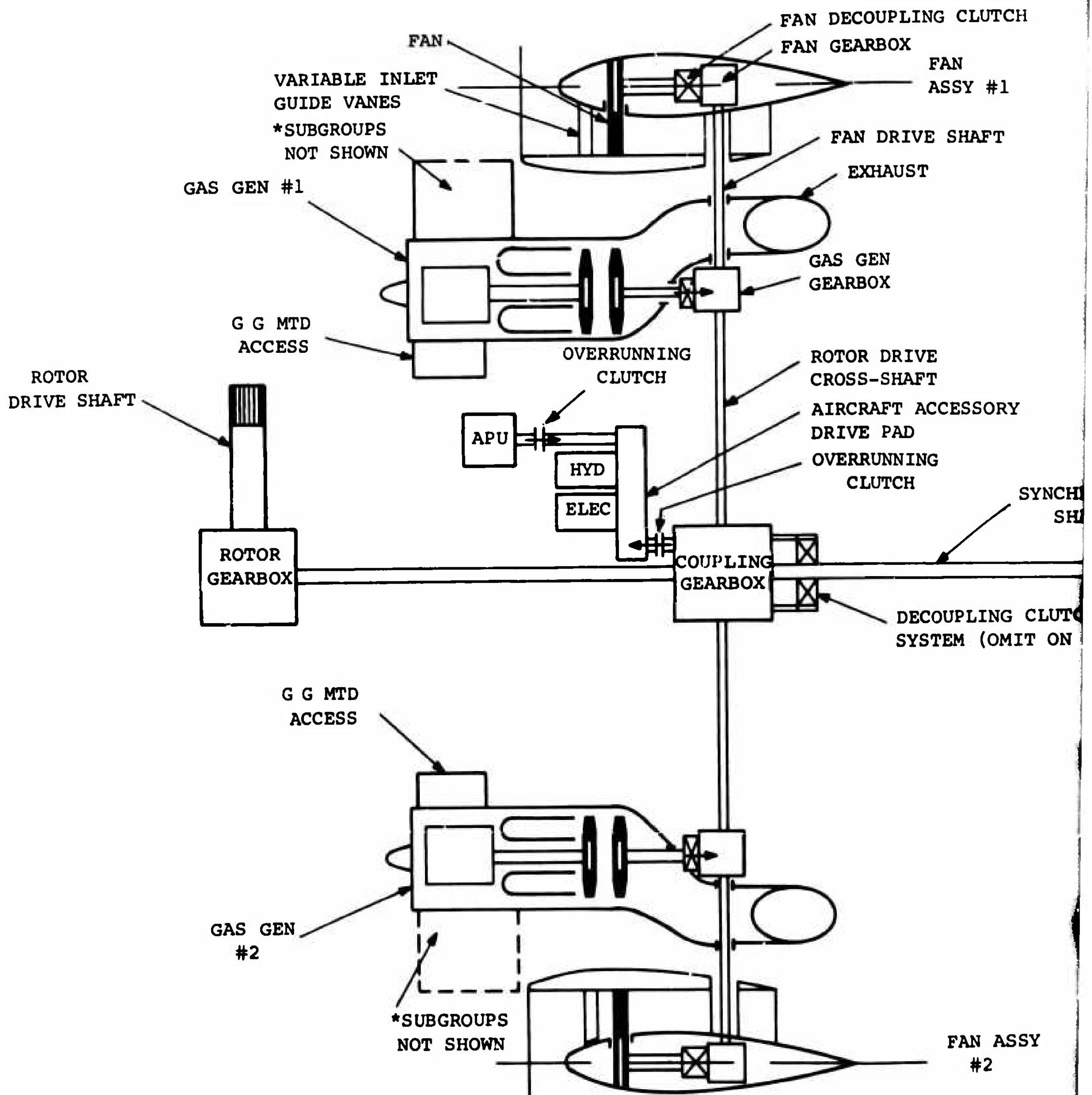
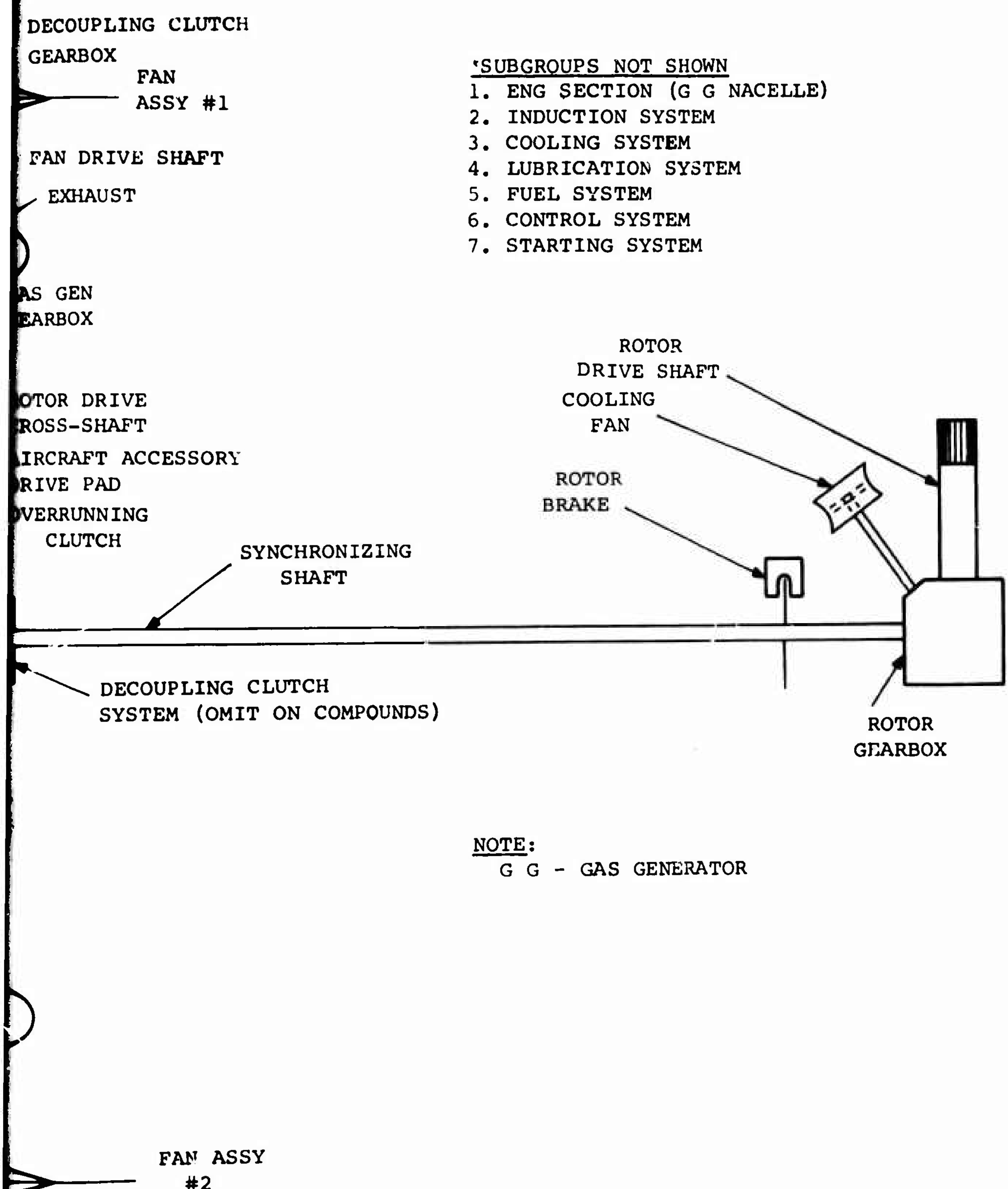


Figure 64. Integrated Lift/Propulsion System Shaf
Remote Cruise Fans - Tandem Rotor Airc
(System 2a), Compound and Composite Ty

A



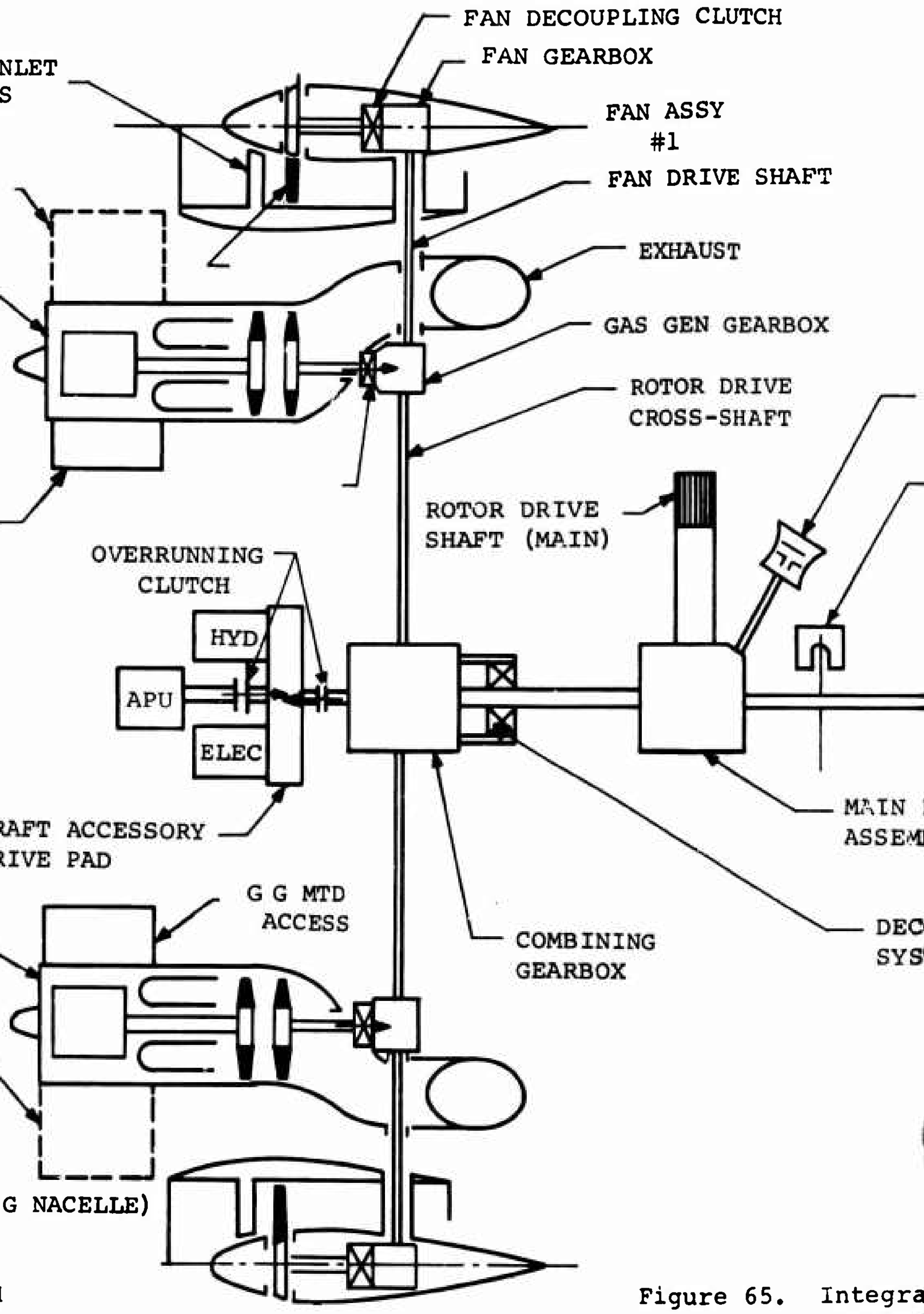
*SUBGROUPS NOT SHOWN

1. ENG SECTION (G G NACELLE)
2. INDUCTION SYSTEM
3. COOLING SYSTEM
4. LUBRICATION SYSTEM
5. FUEL SYSTEM
6. CONTROL SYSTEM
7. STARTING SYSTEM

NOTE:
G G - GAS GENERATOR

/Propulsion System Shaft-Coupled
ns - Tandem Rotor Aircraft
pound and Composite Types

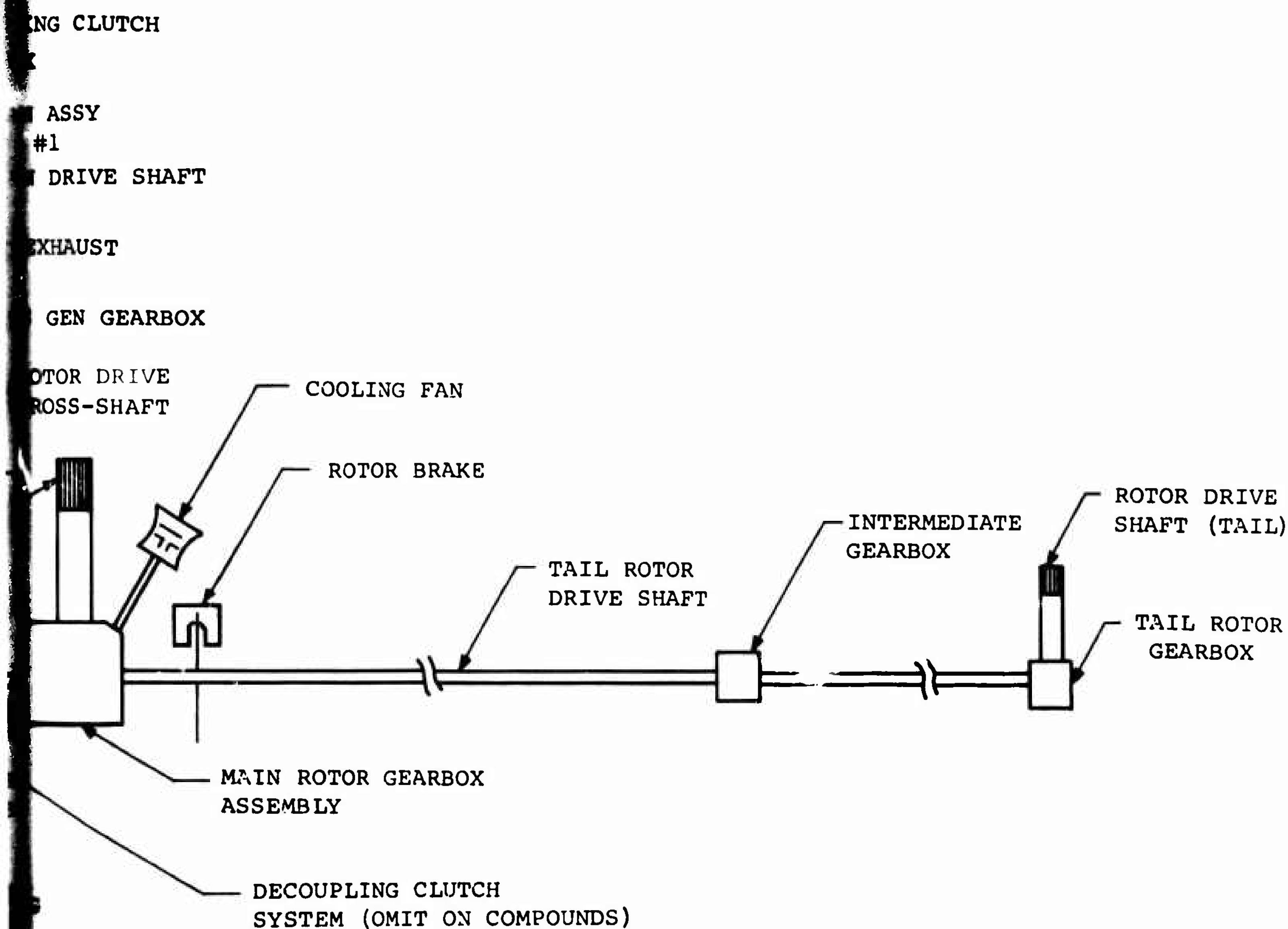
B



*SUBGROUPS NOT SHOWN:

1. ENGINE SECTION (G G NACELLE)
2. INDUCTION SYSTEM
3. COOLING SYSTEM
4. LUBRICATION SYSTEM
5. FUEL SYSTEM
6. CONTROL SYSTEM
7. STARTING SYSTEM

Figure 65. Integra
Coupled
Aircraf
Types



NOTE:
G G - GAS GENERATOR

igure 65. Integrated Lift/Propulsion-Unloaded Shaft-Coupled Remote Cruise Fans - Single Rotor Aircraft (System 2a), Compound and Composite Types

B

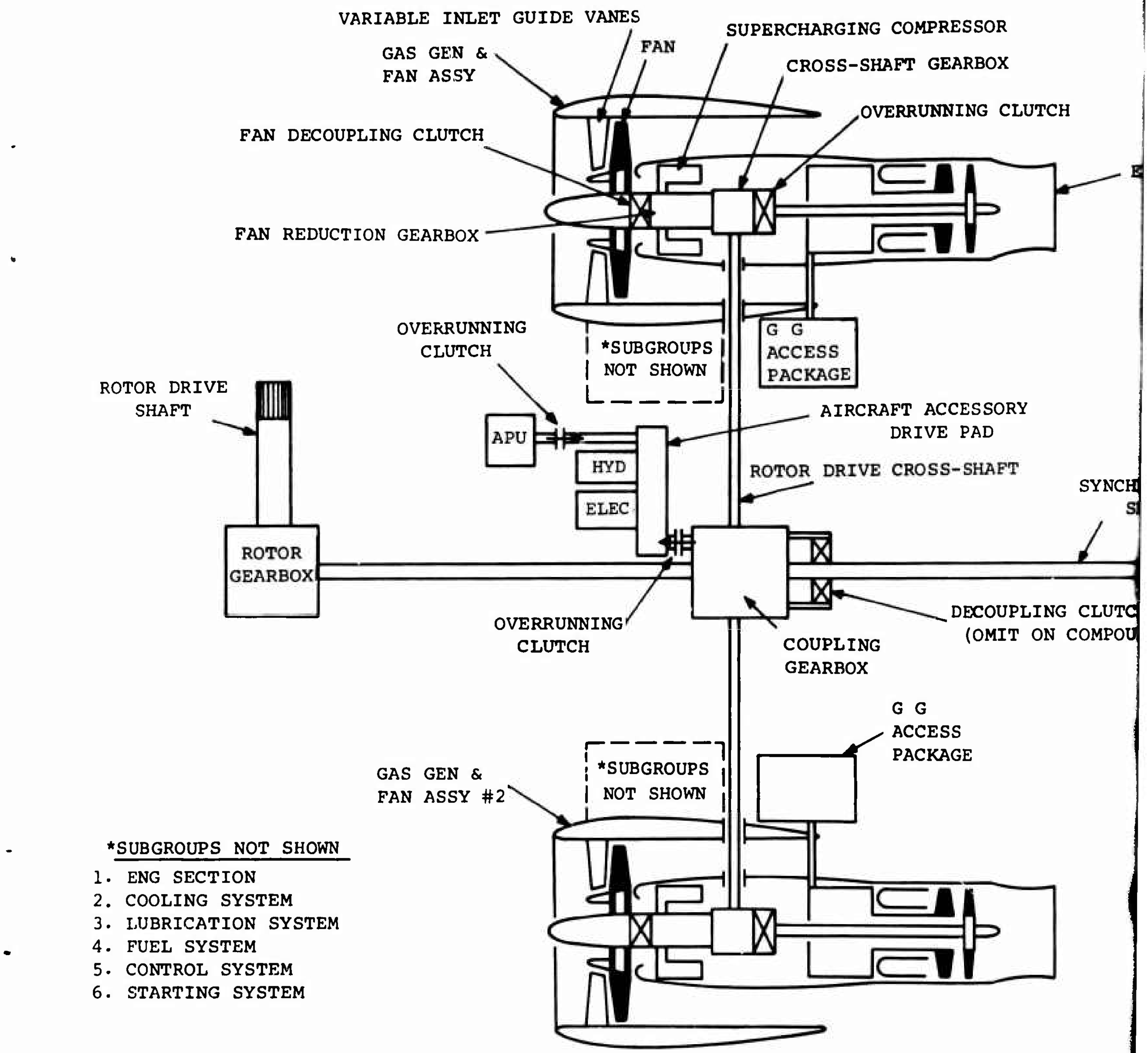
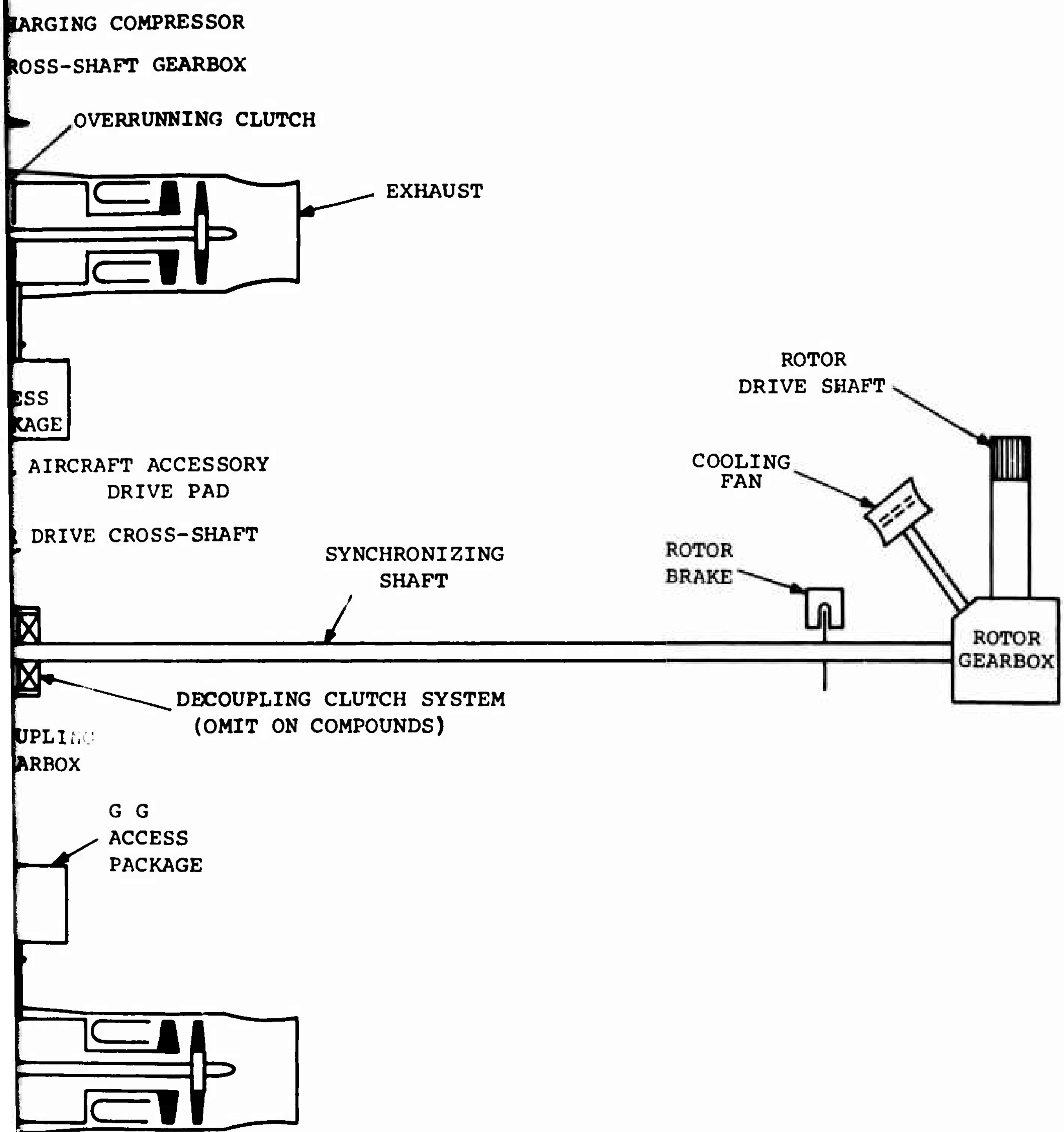


Figure 66. Integrated Lift/Propulsion System Shaft-C
Concentric Axially Driven Cruise Fans - T
Rotor Aircraft (System 2b), Compound and
Composite Types



t/Propulsion System Shaft-Coupled
lly Driven Cruise Fans - Tandem
(System 2b), Compound and

B

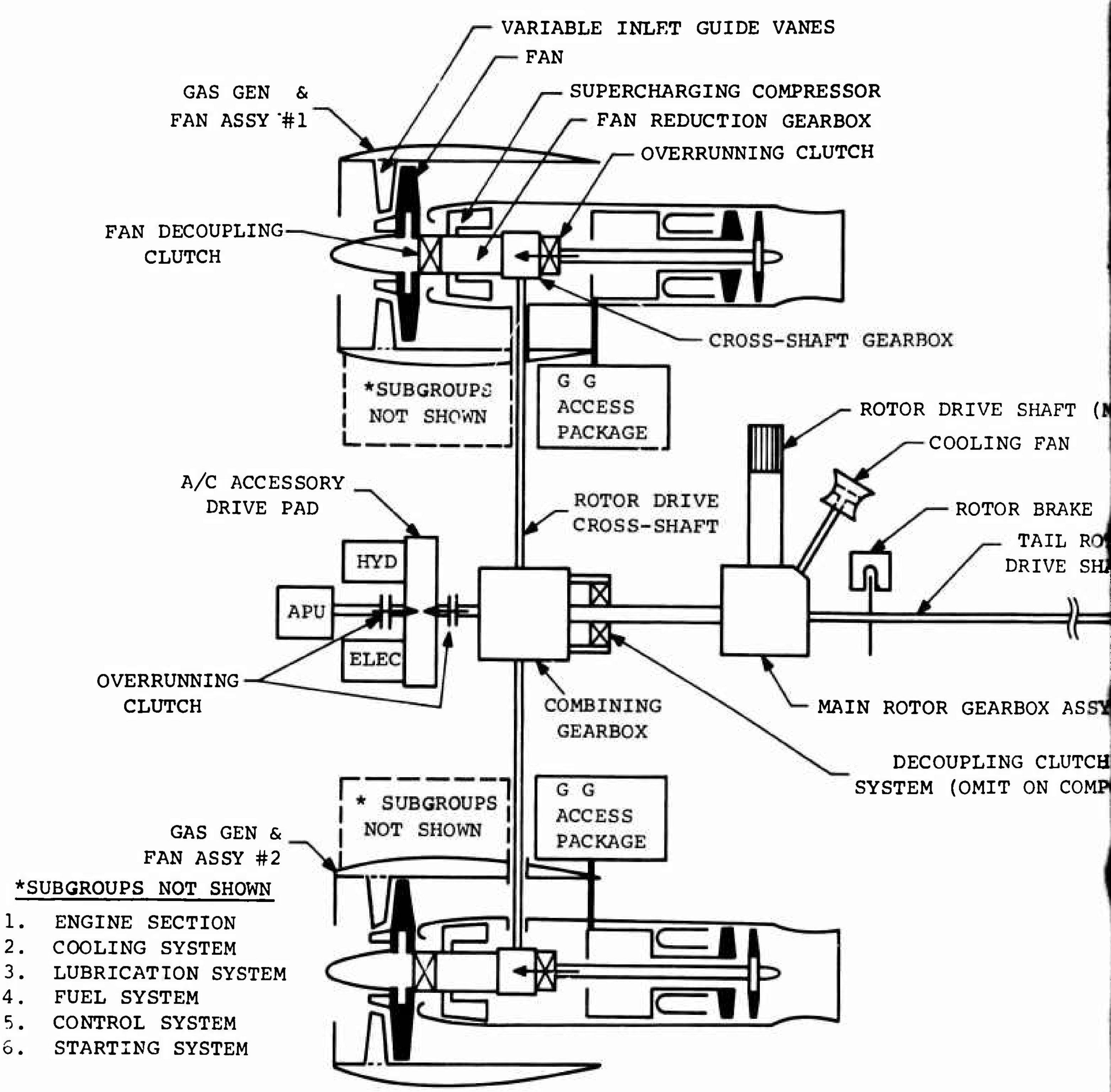
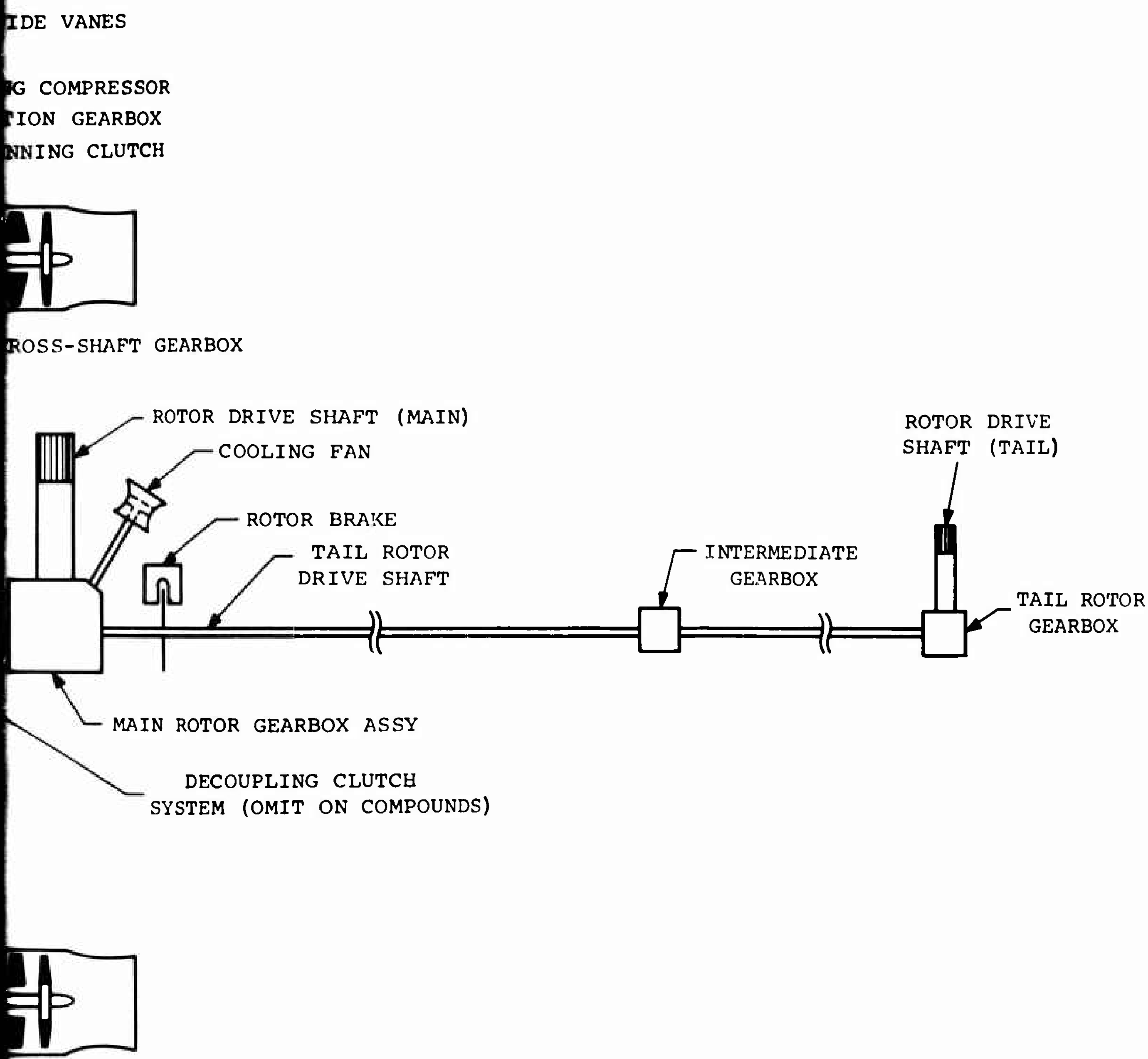
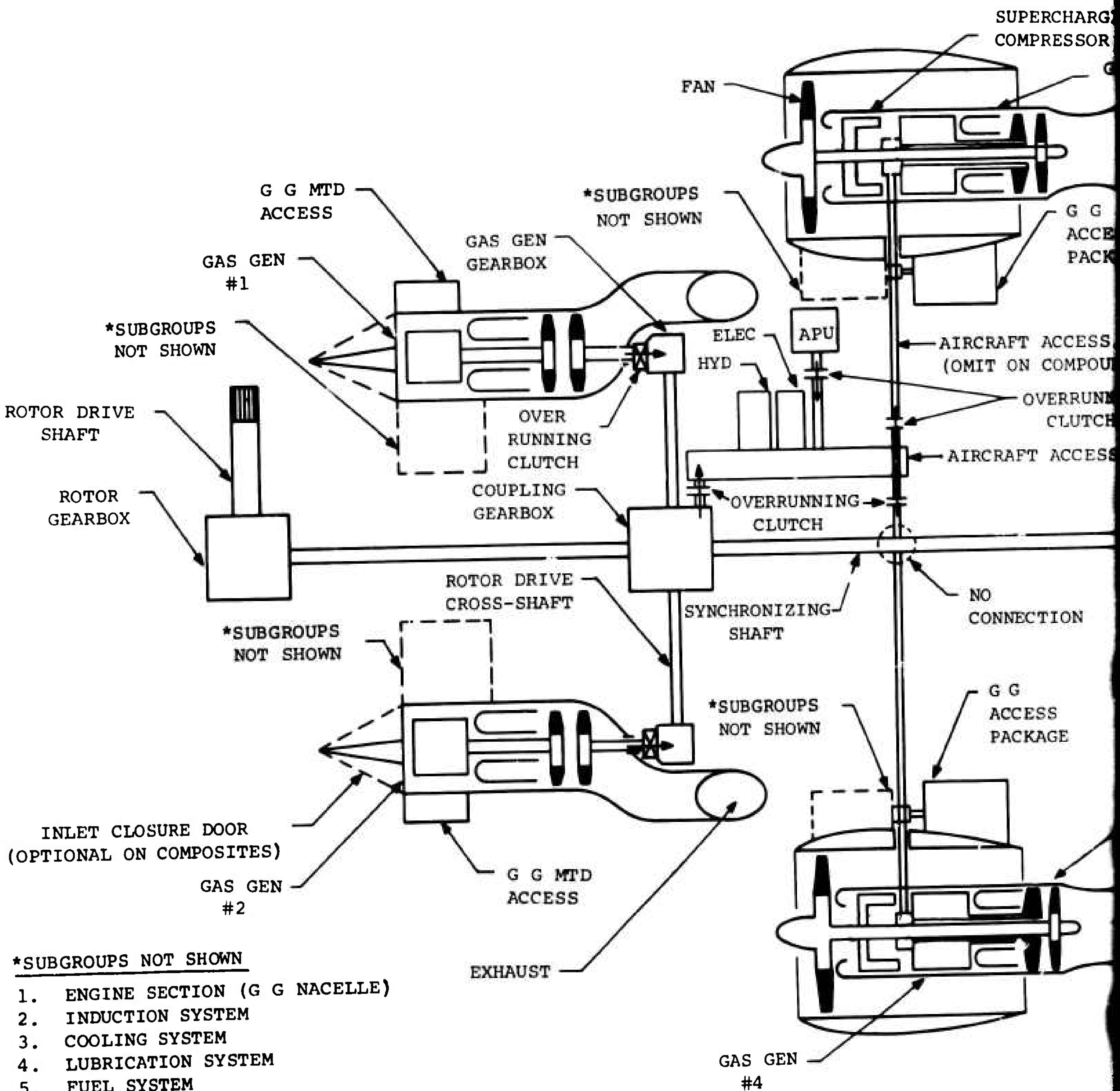


Figure 67. Integrated Lift/Propulsion S
Concentric Axially Driven C
Rotor Aircraft, Compound an



. Integrated Lift/Propulsion System Shaft-Coupled Concentric Axially Driven Cruise Fan (System 2b), Single Rotor Aircraft, Compound and Composite Types



*SUBGROUPS NOT SHOWN

1. ENGINE SECTION (G G NACELLE)
2. INDUCTION SYSTEM
3. COOLING SYSTEM
4. LUBRICATION SYSTEM
5. FUEL SYSTEM
6. CONTROL SYSTEM
7. STARTING SYSTEM

Figure 68. Independent Lift/Propulsion System Conc
Cruise Fans - Tandem Rotor Aircraft (Sy
Compound and Composite Types

A

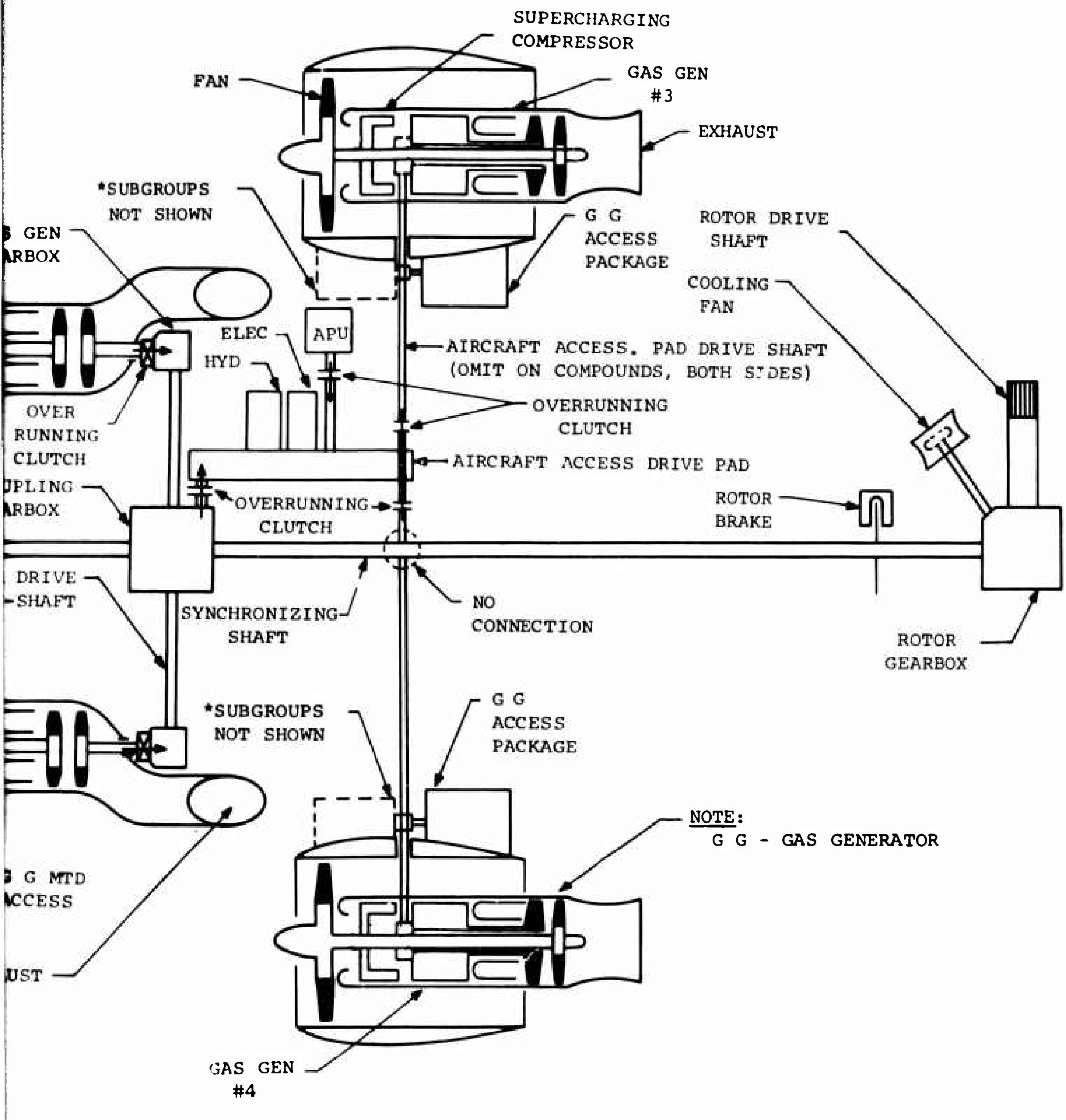


Figure 68. Independent Lift/Propulsion System Concentric Cruise Fans - Tandem Rotor Aircraft (System 3), Compound and Composite Types

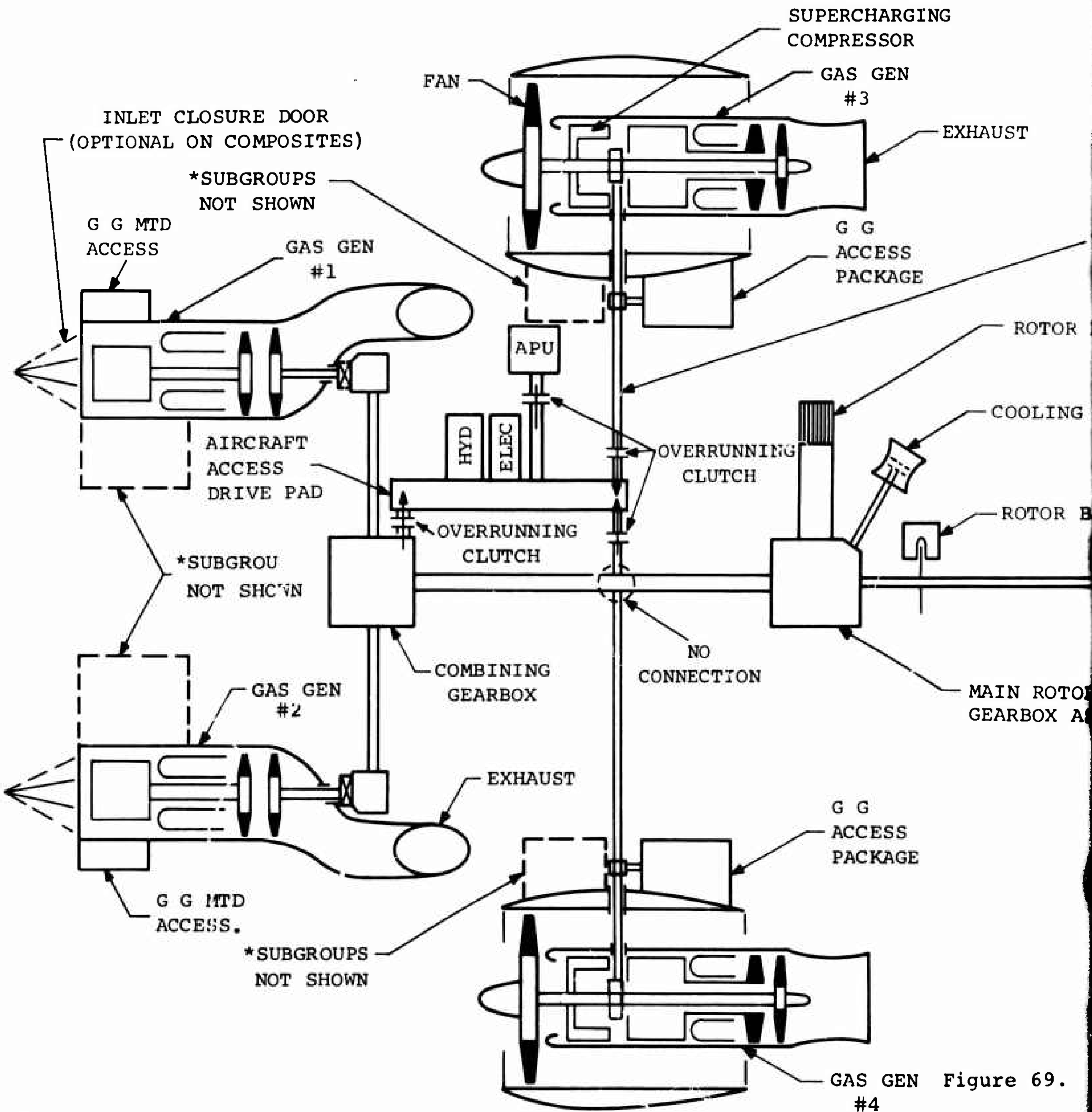

SUPERCHARGING COMPRESSOR
FAN
GAS GEN #3
EXHAUST
INLET CLOSURE DOOR (OPTIONAL ON COMPOSITES)
*SUBGROUPS NOT SHOWN
G G MTD ACCESS
GAS GEN #1
G G ACCESS PACKAGE
APU
ROTOR
COOLING
HYD
ELEC
OVERRUNNING CLUTCH
AIRCRAFT ACCESS DRIVE PAD
OVERRUNNING CLUTCH
ROTOR B
*SUBGROU NOT SHOWN
NO CONNECTION
COMBINING GEARBOX
MAIN ROTO GEARBOX A
GAS GEN #2
EXHAUST
G G ACCESS PACKAGE
G G MTD ACCESS.
*SUBGROUPS NOT SHOWN
GAS GEN #4

Figure 69.

A

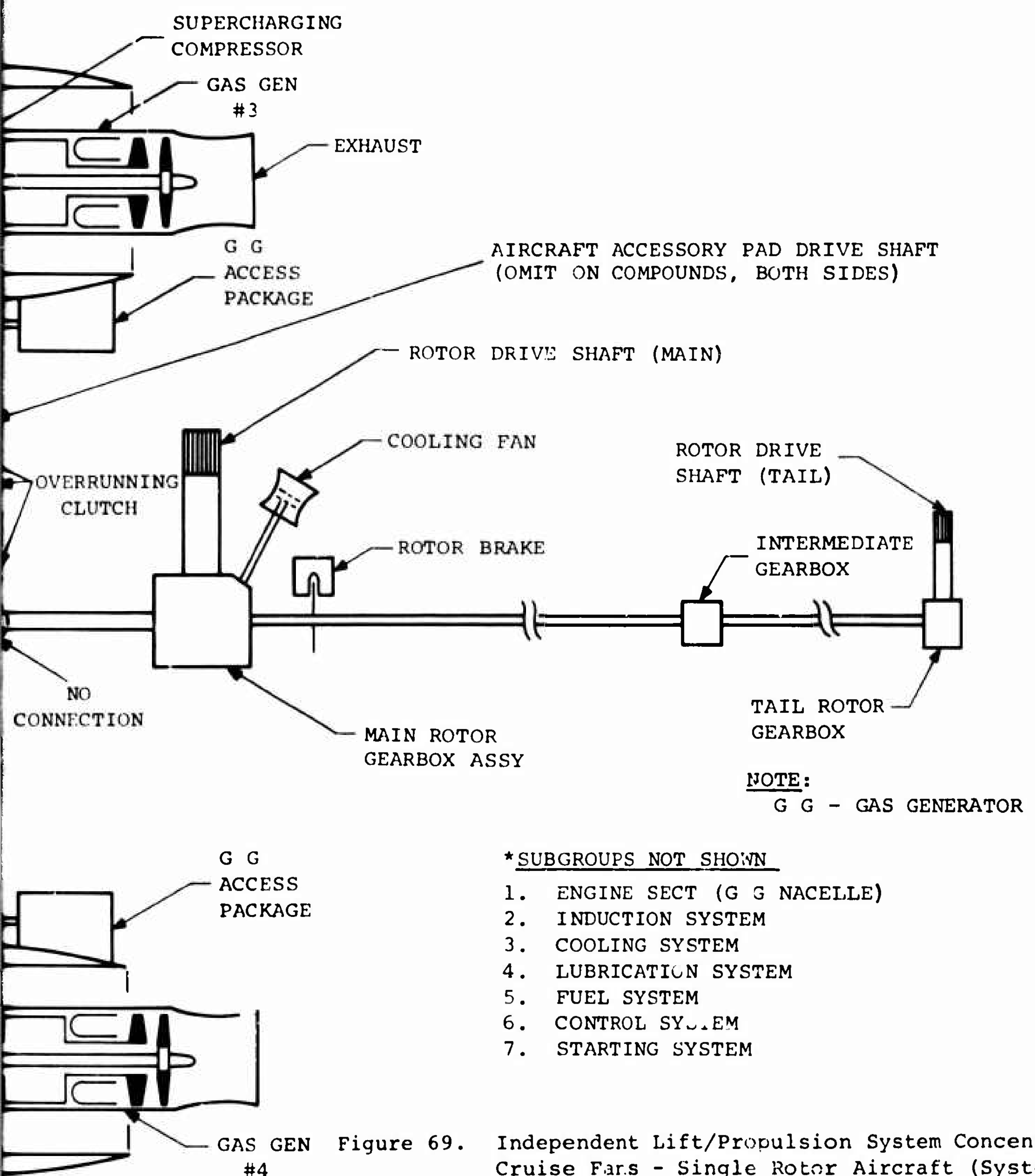
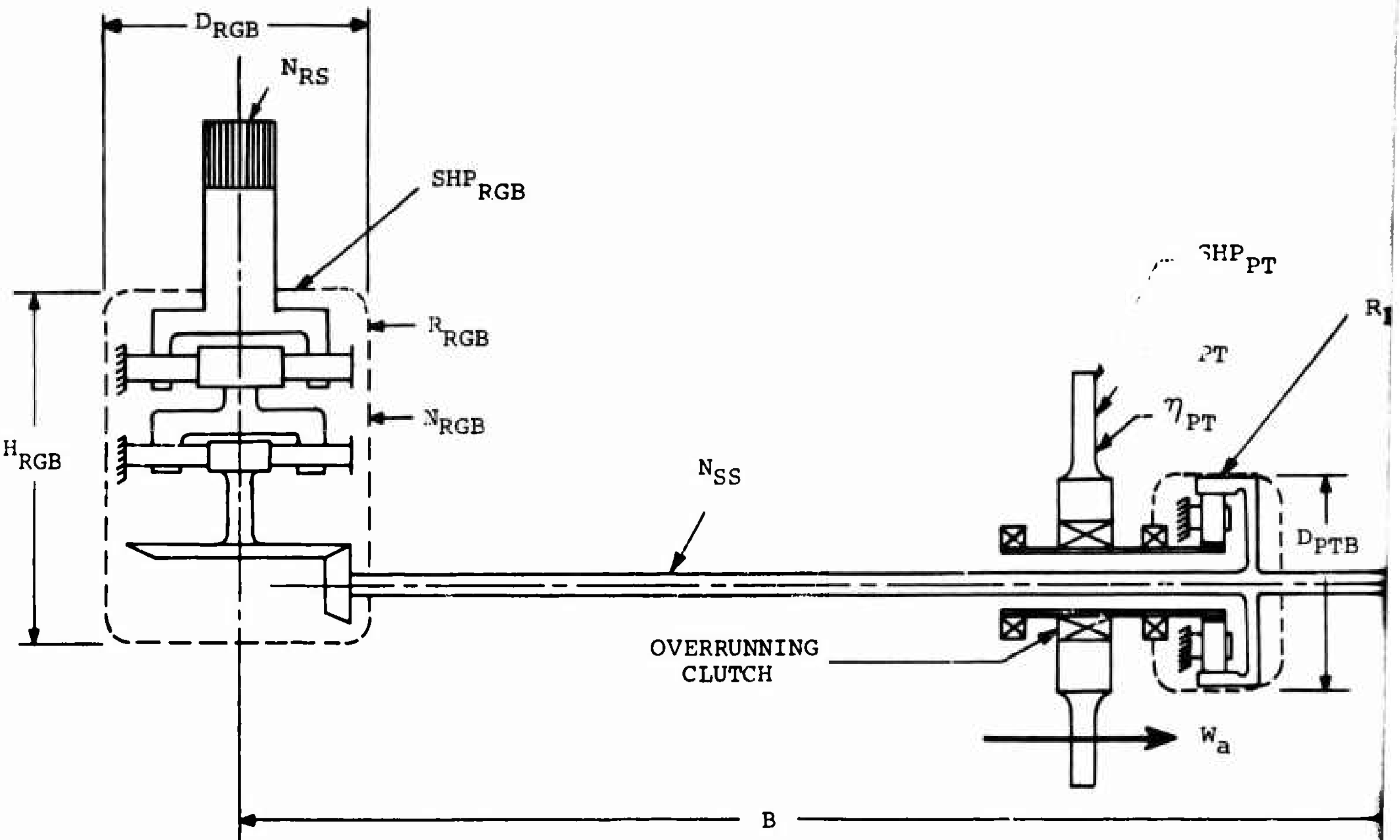


Figure 69. Independent Lift/Propulsion System Concentric Cruise Fans - Single Rotor Aircraft (System 3), Compound and Composite Types

B

NOTE: SYSTEM SPEEDS REFER TO THE HOVER CONDITION ARE MAXIMUM



B = DISTANCE BETWEEN ROTOR CENTERLINES, FEET, = 41.7 + (D-59.1); B= 47.1 MIN.
D = DIAMETER OF ROTOR, FEET
D_{RGB} = DIAMETER OF ROTOR GEARBOX, INCHES, = 0.374 $(D \times SHP_{RGB})^{1/3}$
H_{RGB} = HEIGHT OF ROTOR GEARBOX, INCHES, = 1.8 D_{RGB}
D_{PTB} = DIAMETER OF POWER TURBINE GEARBOX, INCHES, = 0.569 $(SHP_{PT})^{1/3}$
W_a = POWER TURBINE DESIGN AIRFLOW, SL STD, LB/SEC
SHP_{RGB} = SHAFT HORSEPOWER PER ROTOR GEARBOX = 0.595 SHP_{PT}
SHP_{PT} = SHAFT HORSEPOWER AT POWER TURBINE
N_{RS} = SPEED OF ROTOR SHAFT, RPM = $\frac{14310}{D}$ (AT 750 FPS ROTOR TIP SPEED)
N_{SS} = SPEED OF SYNCHRONIZING SHAFT, RPM, = 7155 (CONSTANT AT MAXIMUM SYSTEM SPEED)
N_{PT} = SPEED OF POWER TURBINE SHAFT, RPM = $\frac{120,000}{\sqrt{W_a}}$

Figure 70. Tandem Rotor Gas-Coupled Drive System and Scaling Data for Integrated Lift/P Systems 1a and 1b, Compound and Compo

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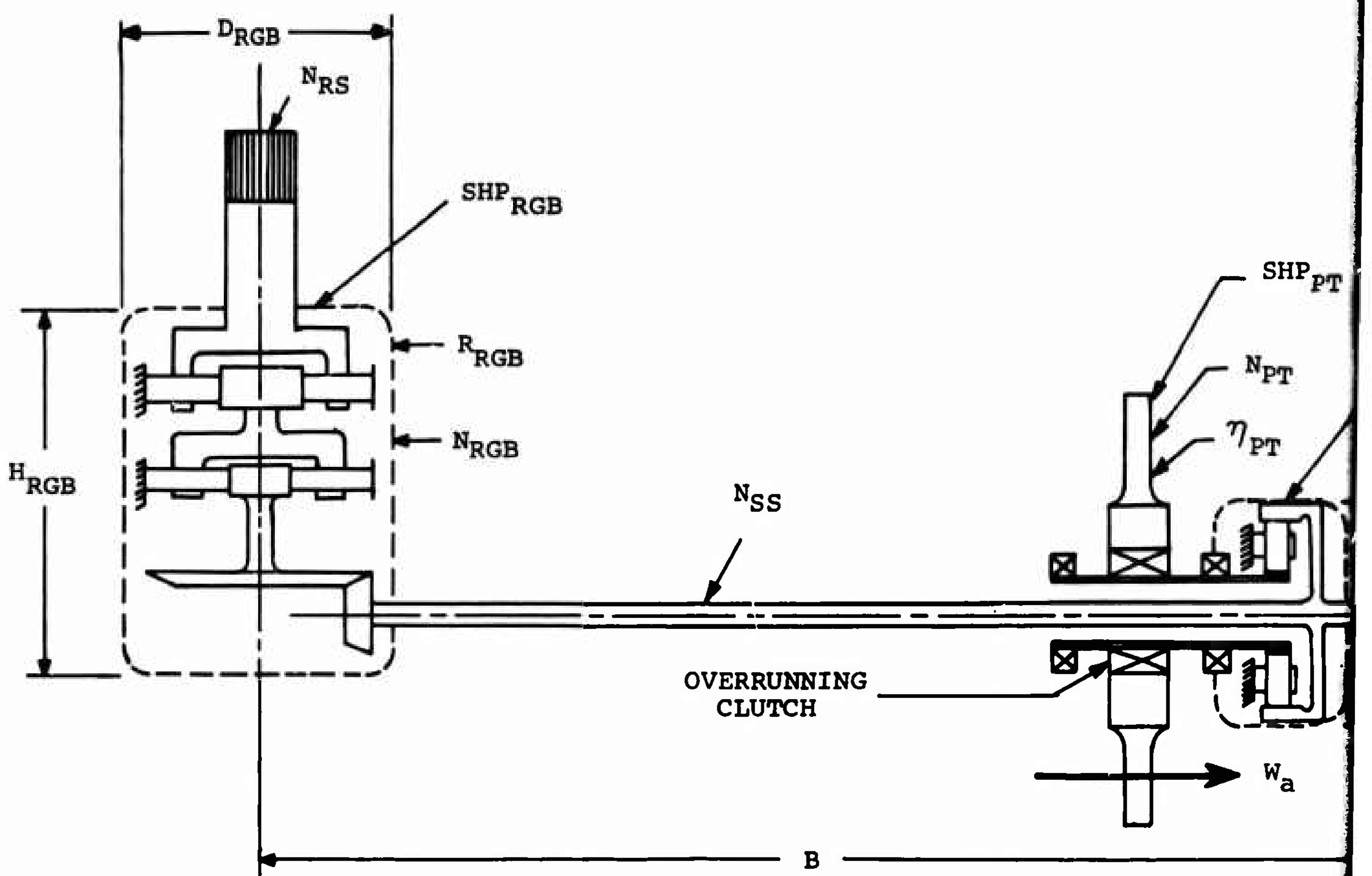
NOTE: SYSTEM SPEEDS REFER TO THE HOVER CO
ARE MAXIMUM

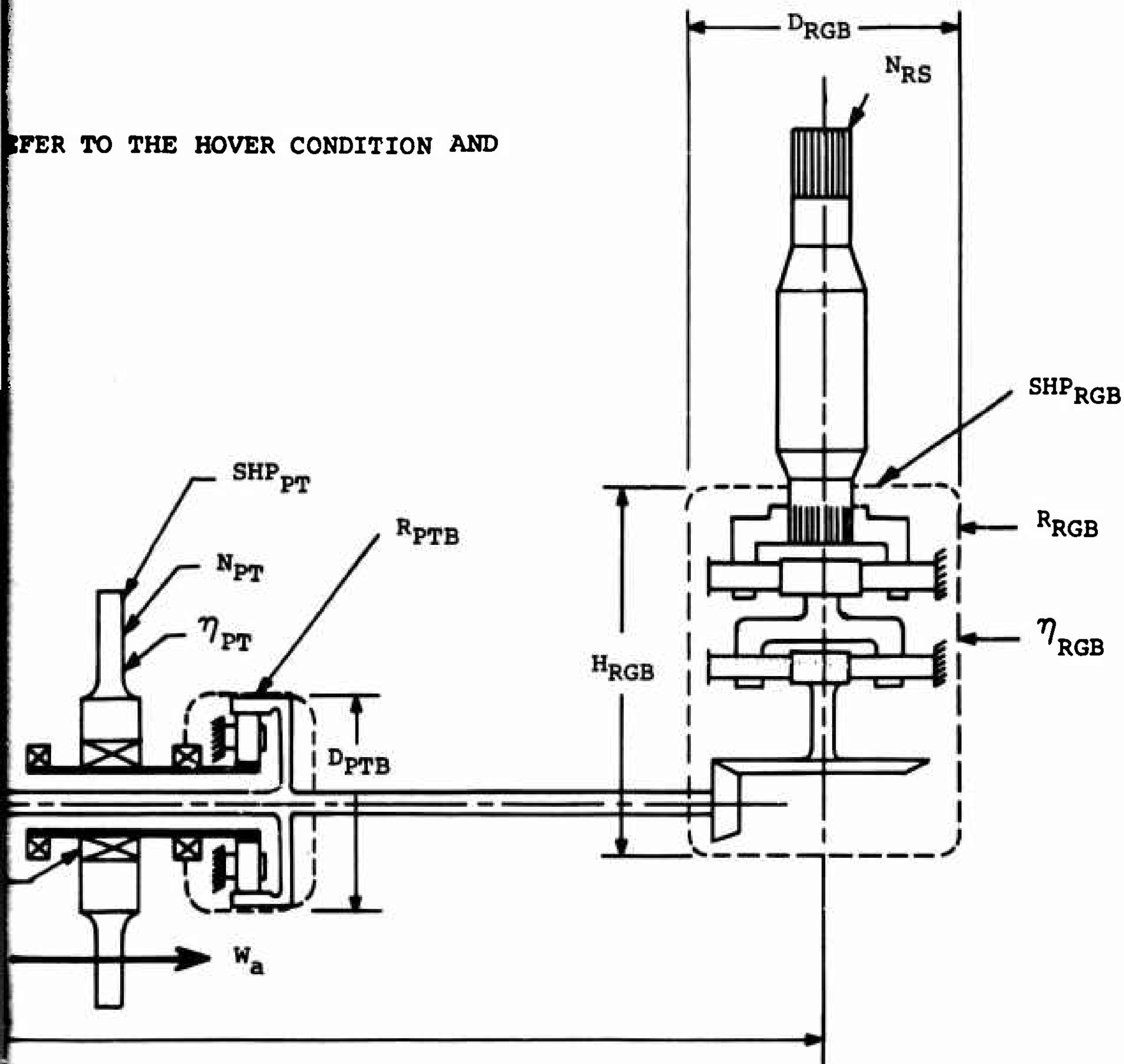


B = DISTANCE BETWEEN ROTOR CENTERLINES, FEET, = 41.7 + (D-59.1); B= 47.1 MIN.
D = DIAMETER OF ROTOR, FEET
D_{RGB} = DIAMETER OF ROTOR GEARBOX, INCHES, = $0.374\ (D \times SHP_{RGB})^{1/3}$
H_{RGB} = HEIGHT OF ROTOR GEARBOX, INCHES, = $1.8\ D_{RGB}$
D_{PTB} = DIAMETER OF POWER TURBINE GEARBOX, INCHES, = $0.569\ (SHP_{PT})^{1/3}$
W_a = POWER TURBINE DESIGN AIRFLOW, SL STD, LB/SEC
SHP_{RGB} = SHAFT HORSEPOWER PER ROTOR GEARBOX = $0.595\ SHP_{PT}$
SHP_{PT} = SHAFT HORSEPOWER AT POWER TURBINE
N_{RS} = SPEED OF ROTOR SHAFT, RPM = $\frac{14310}{D}$ (AT 750 FPS ROTOR TIP SPEED)
N_{SS} = SPEED OF SYNCHRONIZING SHAFT, RPM, = 7155 (CONSTANT AT MAXIMUM SYSTEM SPEED)
N_{PT} = SPEED OF POWER TURBINE SHAFT, RPM = $\frac{120,000}{\sqrt{W_a}}$

Figure 70 . Tandem Rotor Gas-Coupled Drive S
and Scaling Data for Integrated
Systems 1a and 1b, Compound and

A

FER TO THE HOVER CONDITION AND



B= 47.1 MIN.

R_{RGB} = GEAR RATIO OF ROTOR GEARBOX = 0.5D

R_{PTB} = GEAR RATIO OF POWER TURBINE GEARBOX = $\frac{16.8}{\sqrt{W_a}}$

/3

η_{RGB} = EFFICIENCY OF ROTOR GEARBOX = 96.0%

η_{PT} = EFFICIENCY OF POWER TURBINE = 90.0%

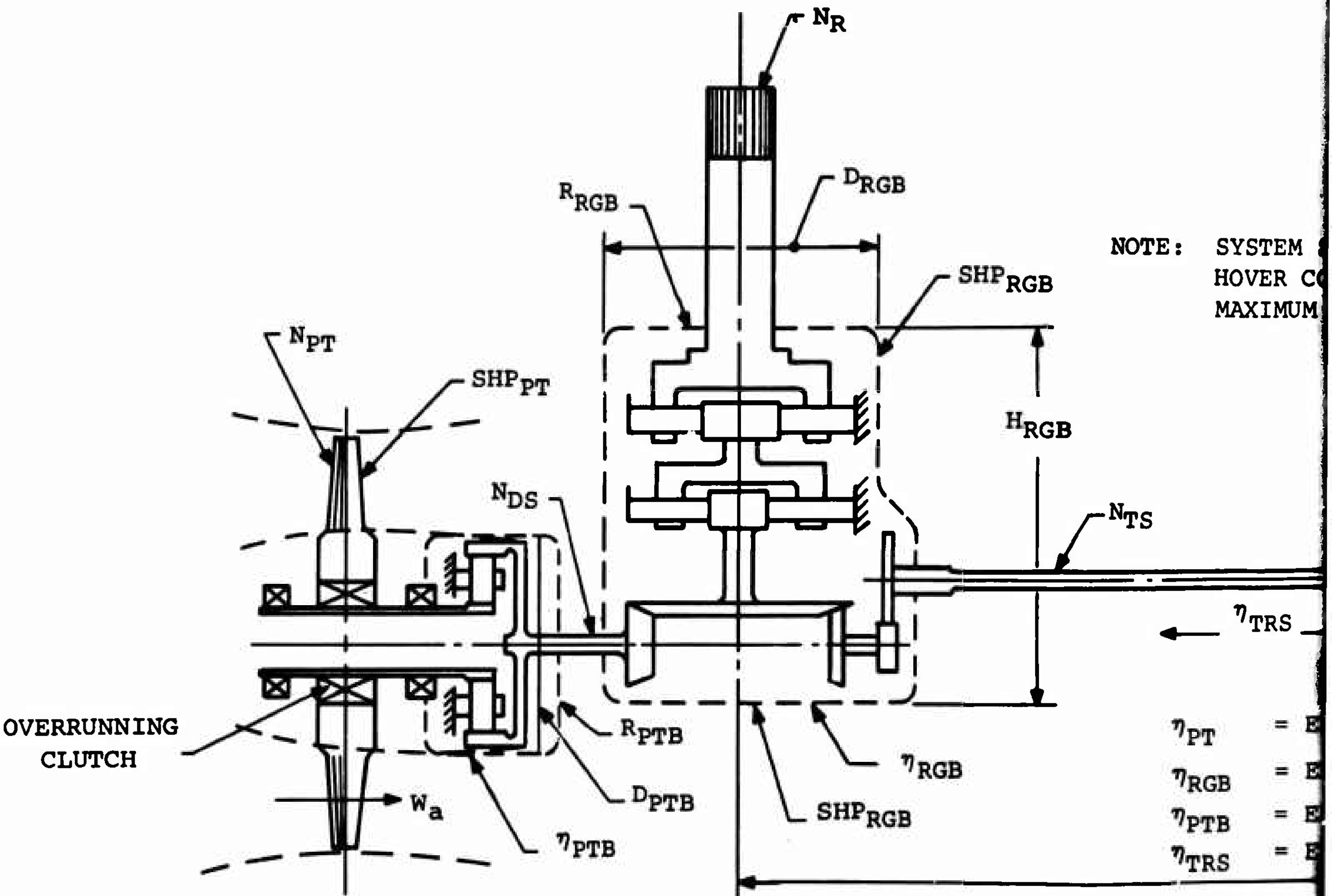
η_{PTB} = EFFICIENCY OF POWER TURBINE GEARBOX = 99.0%

D)

UM SYSTEM SPEED)

as-Coupled Drive System Schematic
ta for Integrated Lift/Propulsion
lb, Compound and Composite Aircraft

B



B = DISTANCE BETWEEN ROTOR CENTERLINES, FEET, $= \frac{(D + D_T)}{2} + 0.5$

D = DIAMETER OF MAIN ROTOR, FEET

D_T = DIAMETER OF TAIL ROTOR, FEET

D_{RGB} = DIAMETER OF MAIN ROTOR GEARBOX, INCHES, $= 0.374\ (D \times SHP_{RGB})^{1/3}$

H_{RGB} = HEIGHT OF MAIN ROTOR GEARBOX, INCHES, $= 1.8\ D_{RGB}$

D_{PTB} = DIAMETER OF POWER TURBINE GEARBOX, INCHES, $= 0.569\ (SHP_{PT})^{1/3}$

W_a = POWER TURBINE DESIGN AIRFLOW, SL STD, LB/SEC

SHP_{RGB} = SHAFT HORSEPOWER AT ROTOR MAIN ROTOR GEARBOX $= 0.90\ SHP_{PT}$

SHP_{PT} = SHAFT HORSEPOWER AT POWER TURBINE

SHP_T = SHAFT HORSEPOWER AT TAIL ROTOR $= 0.10\ SHP_{PT}$

N_R = S

N_{DS} = S

A

N_{TS} = S

N_{PT} =

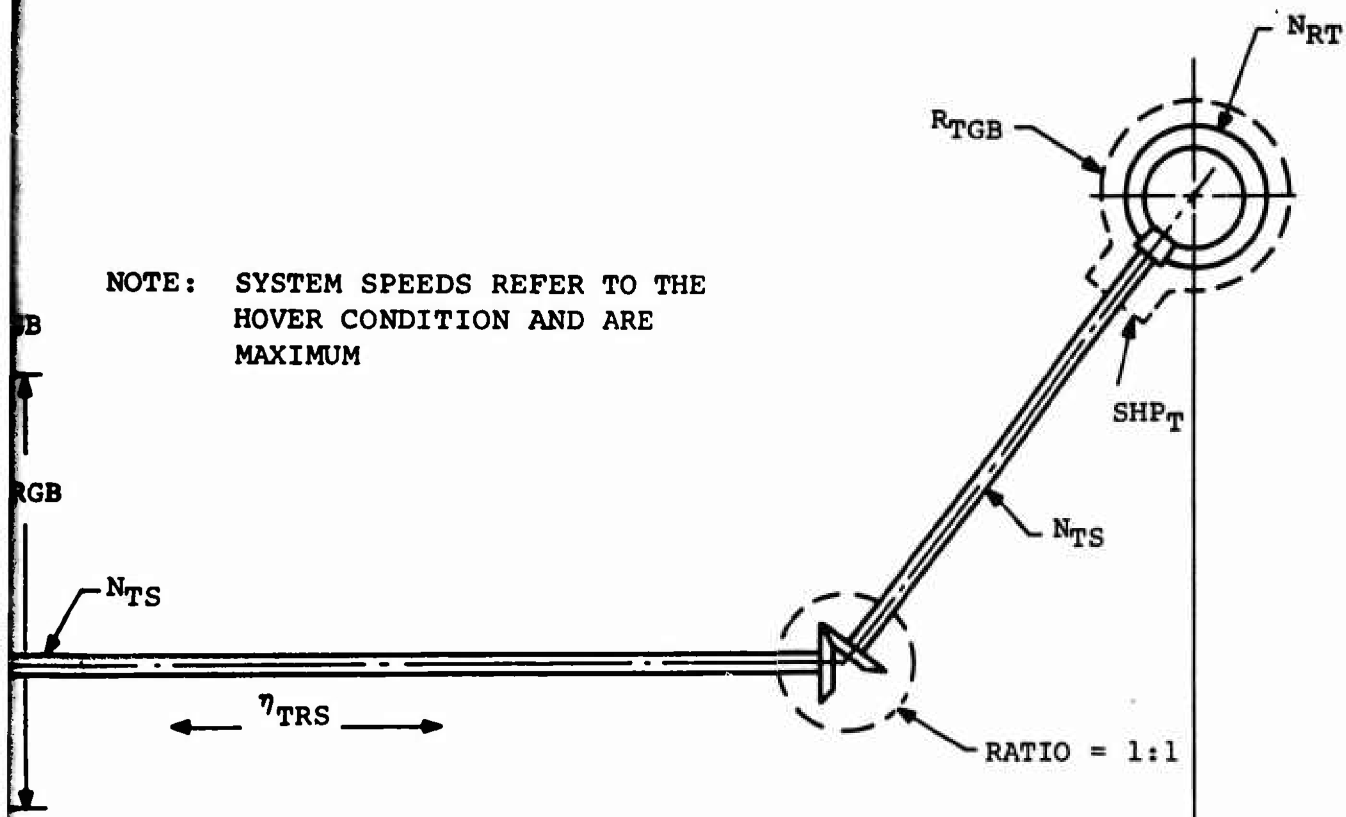
R_{RGB} = G

R_{TGB} = G

R_{PTB} = G

Figure 71. Single Rotor Gas Coupled Drive Systems
1a and 1b, Compound and Composite Aircraft

A



η_{PT} = EFFICIENCY OF POWER TURBINE = 90.0%

η_{RGB} = EFFICIENCY OF MAIN ROTOR GEARBOX = 96.0%

η_{PTB} = EFFICIENCY OF POWER TURBINE GEARBOX = 99.0%

η_{TRS} = EFFICIENCY OF TAIL ROTOR SYSTEM = 96.5%

0.5

N_R = SPEED OF MAIN ROTOR SHAFT, RPM = $\frac{14310}{D}$ (AT 750 FPS ROTOR TIP SPEED)

N_{DS} = SPEED OF MAIN ROTOR GEARBOX INPUT DRIVE SHAFT, RPM = 7155 (CONSTANT AT MAXIMUM SYSTEM SPEED)

$_{RGB})^{1/3}$

N_{TS} = SPEED OF TAIL ROTOR DRIVE SHAFT, RPM = 3000 (CONSTANT AT MAXIMUM SYSTEM SPEED)

$_{PT})^{1/3}$

N_{PT} = SPEED OF POWER TURBINE SHAFT, RPM = $\frac{120,000}{\sqrt{W_a}}$

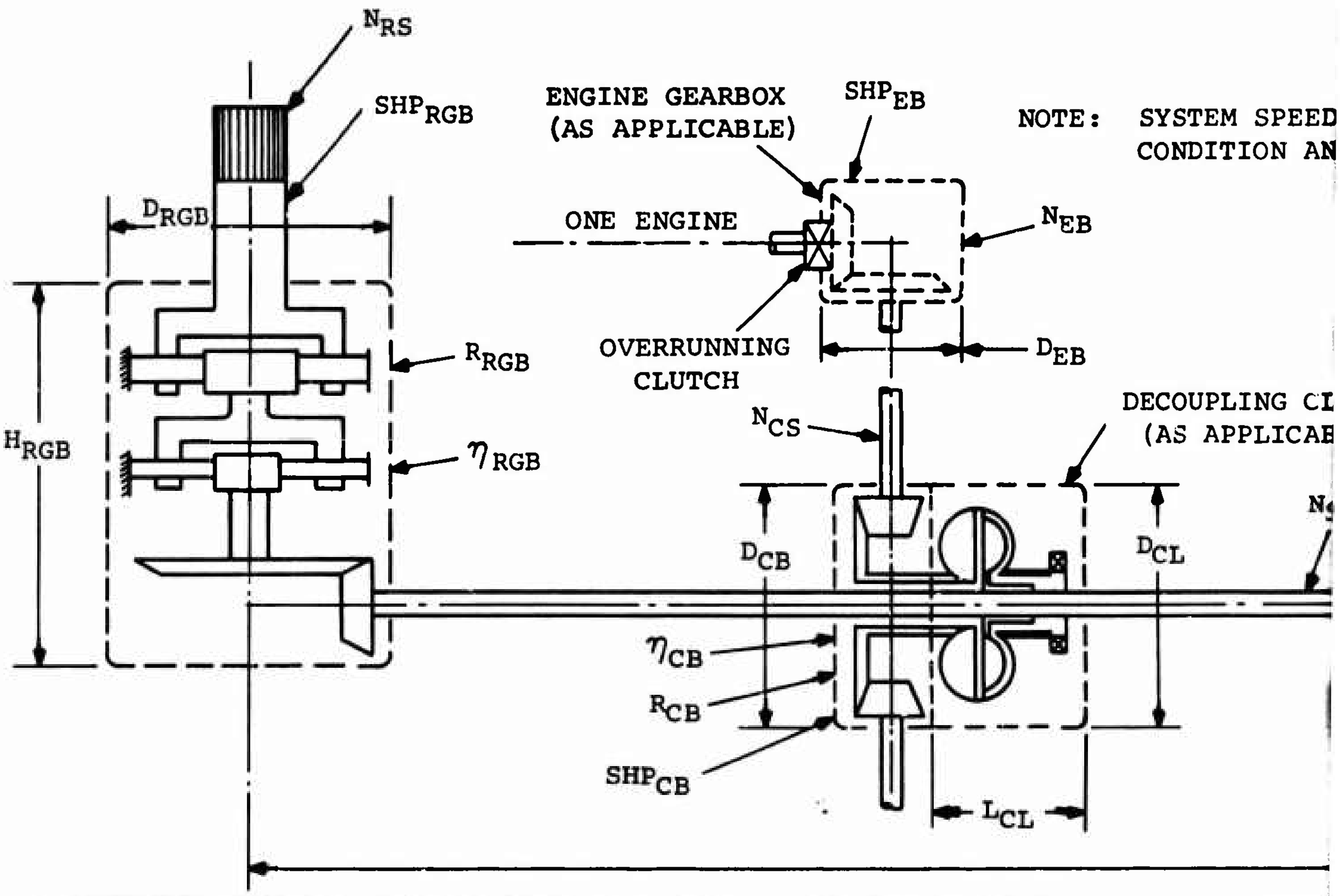
T

R_{RGB} = GEAR RATIO OF MAIN ROTOR GEARBOX = 0.5D

R_{TGB} = GEAR RATIO OF TAIL ROTOR GEARBOX = 0.21 D_T

R_{PTB} = GEAR RATIO OF POWER TURBINE GEARBOX = $\frac{16.8}{\sqrt{W_a}}$

B

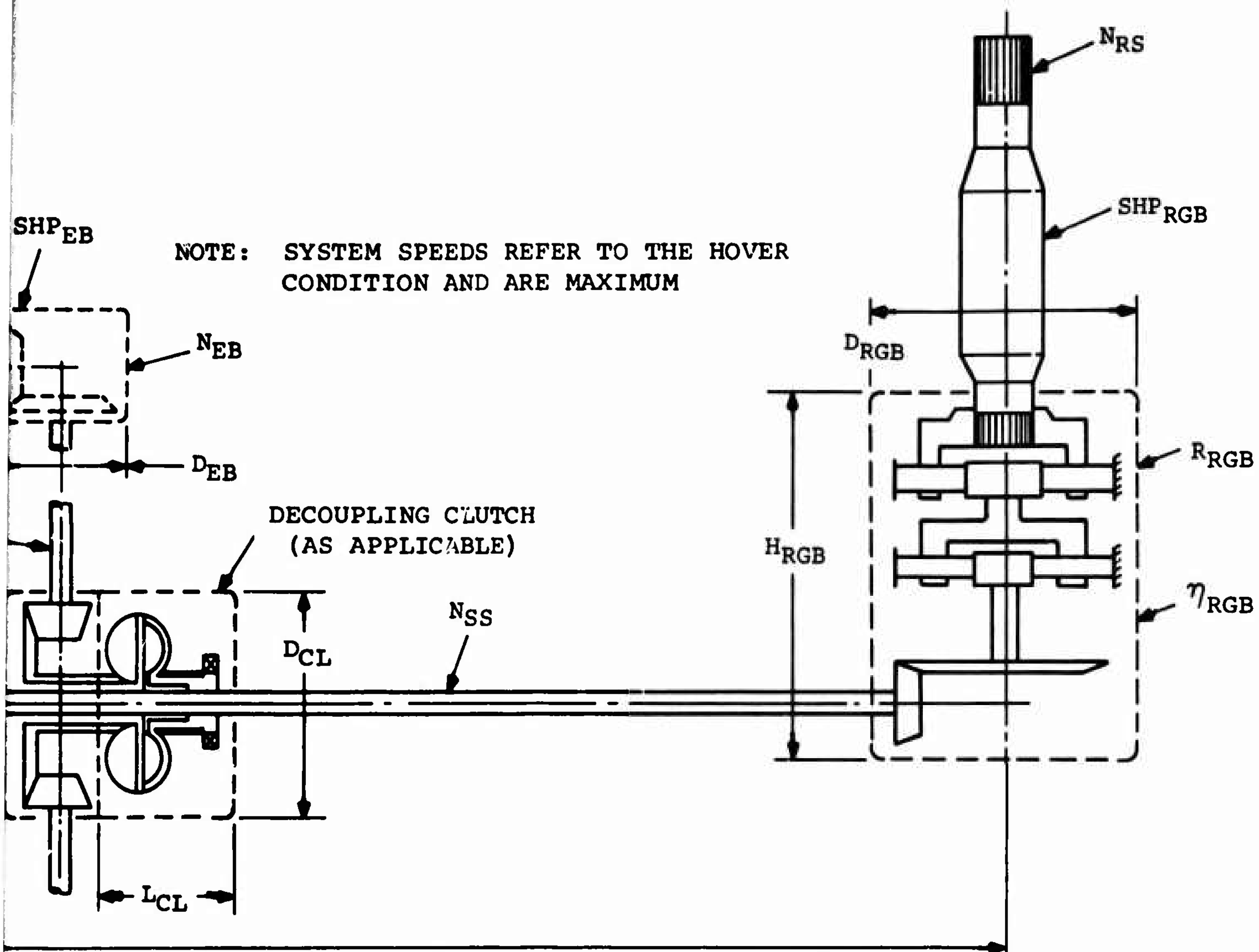


B	=	DISTANCE BETWEEN ROTOR CENTERLINES, FEET, = 41.7 + (D-59.1); B = 47.1 MIN
D	=	DIAMETER OF ROTOR, FEET
D_{RGB}	=	DIAMETER OF ROTOR GEARBOX, INCHES, = $0.374\ (D \times SHP_{RGB})^{1/3}$
H_{RGB}	=	HEIGHT OF ROTOR GEARBOX, INCHES, = $1.8\ D_{RGB}$
D_{CB}	=	DIAMETER OF COMBINING GEARBOX, INCHES, = $0.5\ (SHP_{CB})^{1/3}$
D_{CL}	=	DIAMETER OF DECOUPLING CLUTCH, INCHES, = $0.85\ (SHP_{CB})^{1/3}$
L_{CL}	=	LENGTH OF DECOUPLING CLUTCH, INCHES, = $1.5\ D_{CL}$
D_{EB}	=	DIAMETER OF ENGINE GEARBOX, INCHES, = $0.432\ (SHP_{EB})^{1/3}$
SHP_{RGB}	=	SHAFT HORSEPOWER PER ROTOR GEARBOX = $0.60\ SHP_{CB}$
SHP_{EG}	=	SHAFT HORSEPOWER AT ENGINE GEARBOX
SHP_{CB}	=	SHAFT HORSEPOWER AT COMBINING GEARBOX

N_{RS}	=	SPE
N_{SS}	=	SPE
N_{CS}	=	SPE
N_{RGB}	=	GEA
R_{CB}	=	GEA
η_{RGB}	=	EFF
η_{EB}	=	EFF
η_{CB}	=	EFF

Figure 72. Tandem Rotor Shaft Coupled Drive System Schematic and Scaling Data for Integrated Lift/Propulsion Systems 2a, 2b, and 3, Compound and Composite Aircraft

A



1.7 + (D-59.1); B = 47.1 MIN

x $SHP_{RGB})^{1/3}$

$(SHP_{CB})^{1/3}$

$(SHP_{CB})^{1/3}$

L

$SHP_{EB})^{1/3}$

CB

N_{RS} = SPEED OF ROTOR SHAFT, RPM = $\frac{14310}{D}$ (AT 750 FPS ROTOR TIP SPEED)

N_{SS} = SPEED OF SYNCHRONIZING SHAFT, RPM, = 7155 (CONSTANT AT MAXIMUM SYSTEM SPEED)

N_{CS} = SPEED OF CROSS SHAFT, RPM, = 11,000 (CONSTANT AT MAXIMUM SYSTEM SPEED)

N_{RGB} = GEAR RATIO OF ROTOR GEARBOX = 0.5D

R_{CB} = GEAR RATIO OF COMBINING GEARBOX = 1.537:1 (CONSTANT)

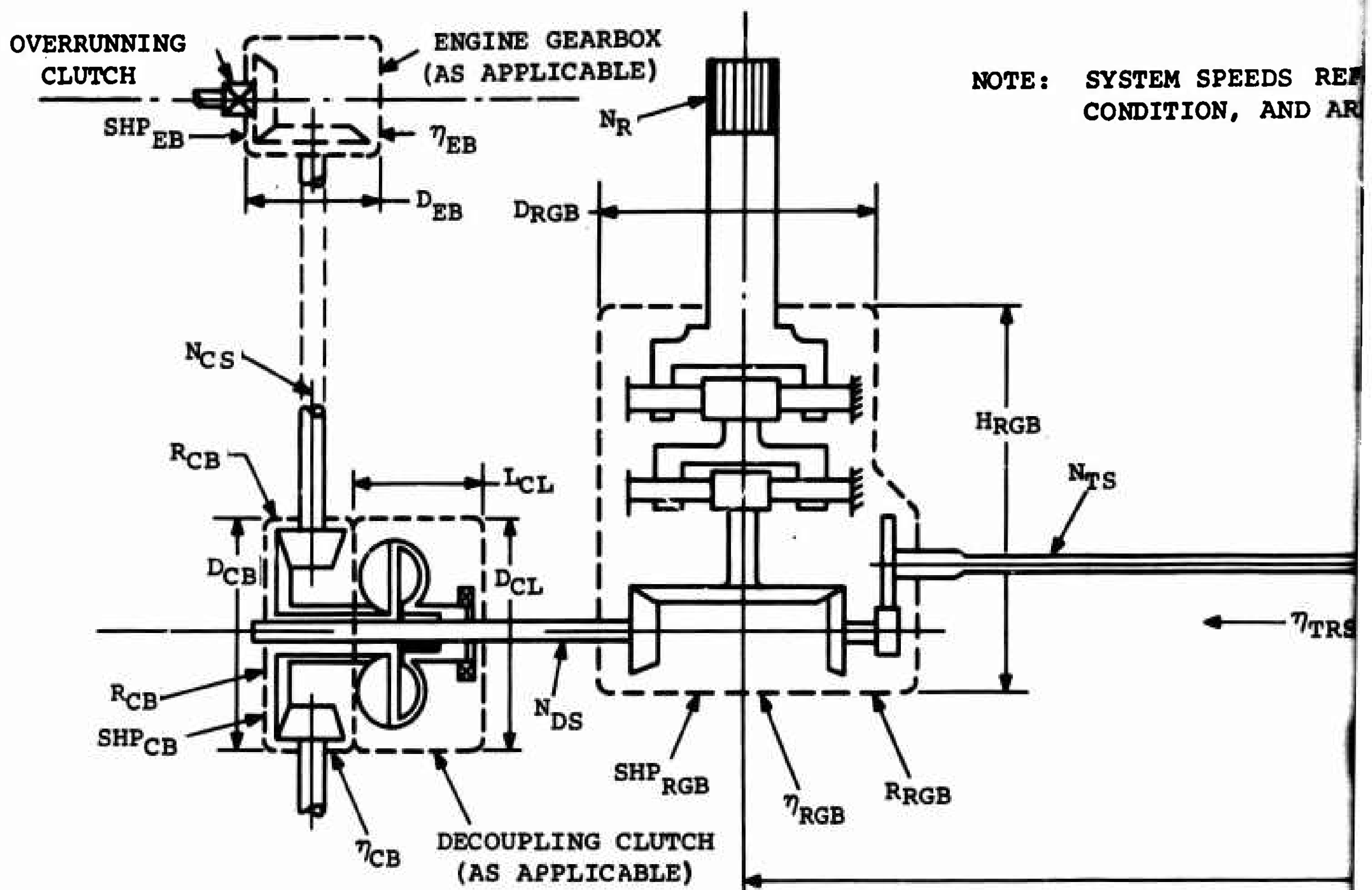
η_{RGB} = EFFICIENCY OF ROTOR GEARBOX = 96.0%

η_{EB} = EFFICIENCY OF ENGINE GEARBOX = 99.0%

η_{CB} = EFFICIENCY OF COMBINING GEARBOX = 99.0%

ematic
sion Systems
aft

B



B = DISTANCE BETWEEN ROTOR CENTERLINES, FEET, $= \frac{(D + D_T)}{2} + 0.5$

D = DIAMETER OF MAIN ROTOR, FEET

D_T = DIAMETER OF TAIL ROTOR, FEET

D_{RGB} = DIAMETER OF MAIN ROTOR GEARBOX, INCHES, $= 0.374\ (D \times SHP_{RGB})^{1/3}$

H_{RGB} = HEIGHT OF MAIN ROTOR GEARBOX, INCHES, $= 1.8\ D_{RGB}$

D_{CB} = DIAMETER OF COMBINING GEARBOX, INCHES, $= 0.5\ (SHP_{CB})^{1/3}$

D_{EB} = DIAMETER OF ENGINE GEARBOX, INCHES, $= 0.432\ (SHP_{EB})^{1/3}$

D_{CL} = DIAMETER OF DECOUPLING CLUTCH, INCHES, $= 0.85\ (SHP_{CB})^{1/3}$

L_{CL} = LENGTH OF DECOUPLING CLUTCH, INCHES, $= 1.5\ D_{CL}$

SHP_{RGB} = SHAFT HORSEPOWER AT MAIN ROTOR GEARBOX $= 0.90\ SHP_{CB}$

SHP_T = SHAFT HORSEPOWER AT TAIL ROTOR $= 0.10\ SHP_{CB}$

SHP_{CB} = SHAFT HORSEPOWER AT COMBINING GEARBOX

SHP_{EB} = SHAFT HORSEPOWER AT ENGINE GEARBOX

N_{CS} = SPE

N_{DS} = SPE
(CO

N_{TS} = SPE

NR_T = SPE

R_{RGB} = GE

R_{TGB} = GEA

R_{CB} = GEA

η_{EB} = EFF

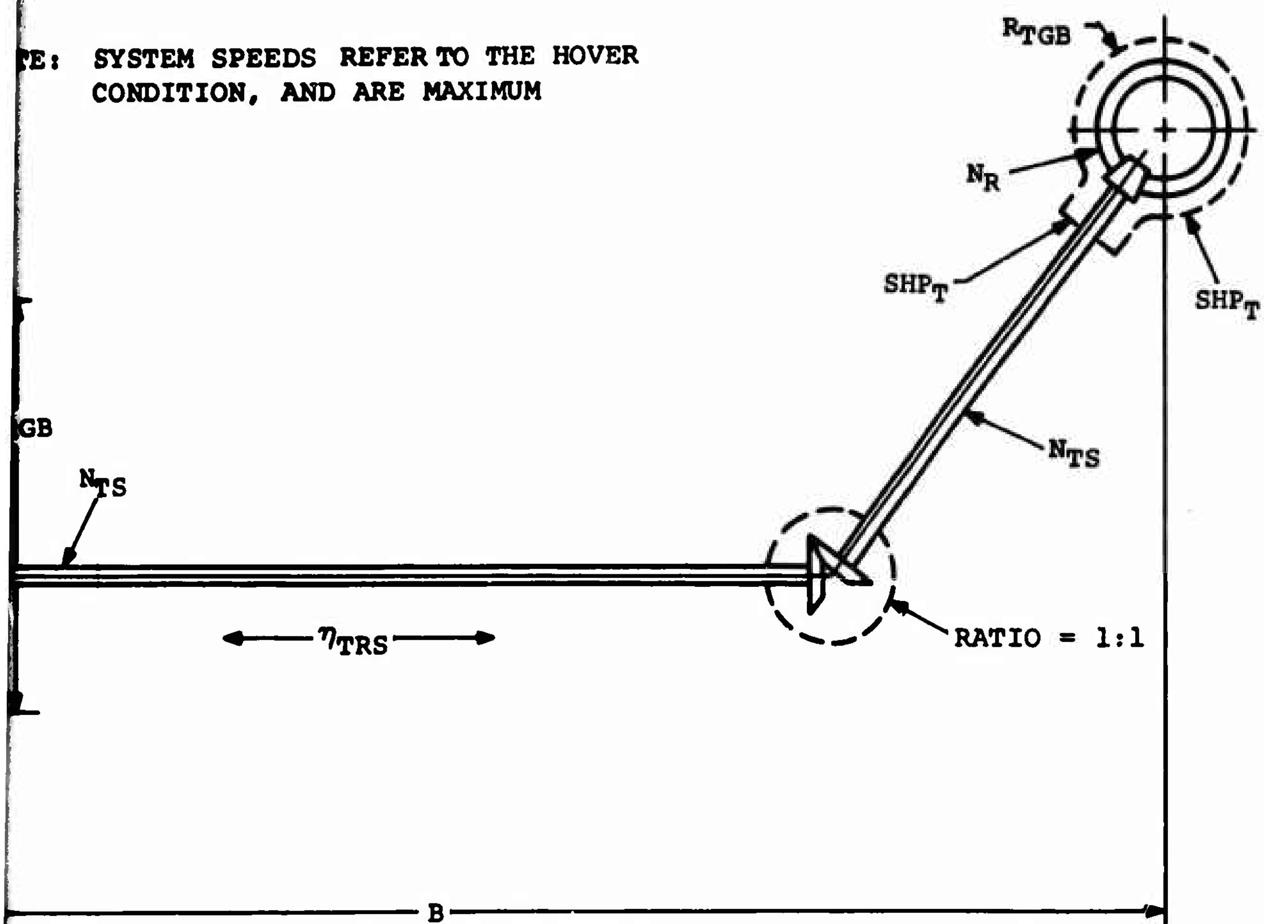
η_{CB} = EFF

η_{RGB} = EFF

η_{TRS} = EFF

Figure 73. Single Rotor Shaft-Coupl
and Scaling Data for Int
Systems 2a, 2b, and 3, C

A



B) 1/3

N_{CS} = SPEED OF CROSS-SHAFT, RPM = 11,000 (CONSTANT AT MAXIMUM SYSTEM SPEED)

N_{DS} = SPEED OF MAIN ROTOR GEARBOX INPUT DRIVE SHAFT,RPM, = 7155 (CONSTANCE AT MAXIMUM SYSTEM SPEED)

N_{TS} = SPEED OF TAIL ROTOR DRIVE SHAFT,RPM, = 3000 (CONSTANT AT MAXIMUM SYSTEM SPEED)

NR_T = SPEED OF TAIL ROTOR, RPM = $\frac{14310}{D_T}$ (AT 750 RPS MAIN ROTOR TIP SPEED)

R_{RGB} = GEAR RATIO OF MAIN ROTOR GEARBOX = 0.5D

R_{TGB} = GEAR RATIO OF TAIL ROTOR GEARBOX = 0.21 D_T

R_{CB} = GEAR RATIO OF COMBINING GEARBOX = 1.537:1 (CONSTANT)

η_{EB} = EFFICIENCY OF ENGINE GEARBOX = 99.0%

η_{CB} = EFFICIENCY OF COMBINING GEARBOX = 99.0%

η_{RGB} = EFFICIENCY OF MAIN ROTOR GEARBOX = 96.0%

η_{TRS} = EFFICIENCY OF TAIL ROTOR SYSTEM = 96.5%

gle Rotor Shaft-Coupled Drive System Schematic
Scaling Data for Integrated Lift/Propulsion
tems 2a, 2b, and 3, Compound and Composite Aircraft

PROPULSION SYSTEM AND AIRFRAME INTEGRATION

CONFIGURATION DESIGN INVESTIGATION

This section discusses the basic airframe-propulsion system integration factors that appeared to be most important from the design viewpoint during general layout investigations. These design studies were made prior to the determination of the size and weight of the optimum aircraft of each type. Layouts of all airframe-propulsion combinations were made with an assumed 35,000-pound design gross weight to represent the mid-range of the weight spectrum employed in scaling. In general, the difficulties of location and relative arrangement of propulsion components of all systems were no greater than those normally encountered in VTOL vehicle design. Certain factors causing design installation difficulty became apparent, however; and, as would be expected, configuration problems of increasing complexity were associated with propulsion systems having the largest number of separate major components, as well as with those of the gas-coupled type. Systems employing gas generators remote from cruise fan assemblies, whether integrated or independent, presented an increased number of packaging problems and increased the amount of aircraft protuberances. In addition, these types of systems more severely limited the choice of available locations and the flexibility of movement for purposes of aircraft balance. The gas-coupled systems incorporate a relatively large, common power-turbine package that became a problem from aerodynamic fairing, structural, and accessibility points of view. Flexibility of gas-coupled system component location was limited by considerations of desired minimum duct lengths from a vulnerability standpoint, and the proper integration of ducting with structure from both space and heat insulation considerations restricted potential design approaches. The least overall difficult installation was encountered with the integrated shaft-coupled concentric (convertible) cruise-fan system, since a minimum number of components was involved; the largest of the components was necessarily located well outboard of the basic skin lines of the aircraft.

Tandem Rotor Gas-Coupled Systems 1a and 1b

Several potential locations of gas-coupled systems on tandem rotor aircraft were investigated. These included long-coupled configurations, with extended lengths of gas ducting between major components, and short-coupled arrangements, where all parts

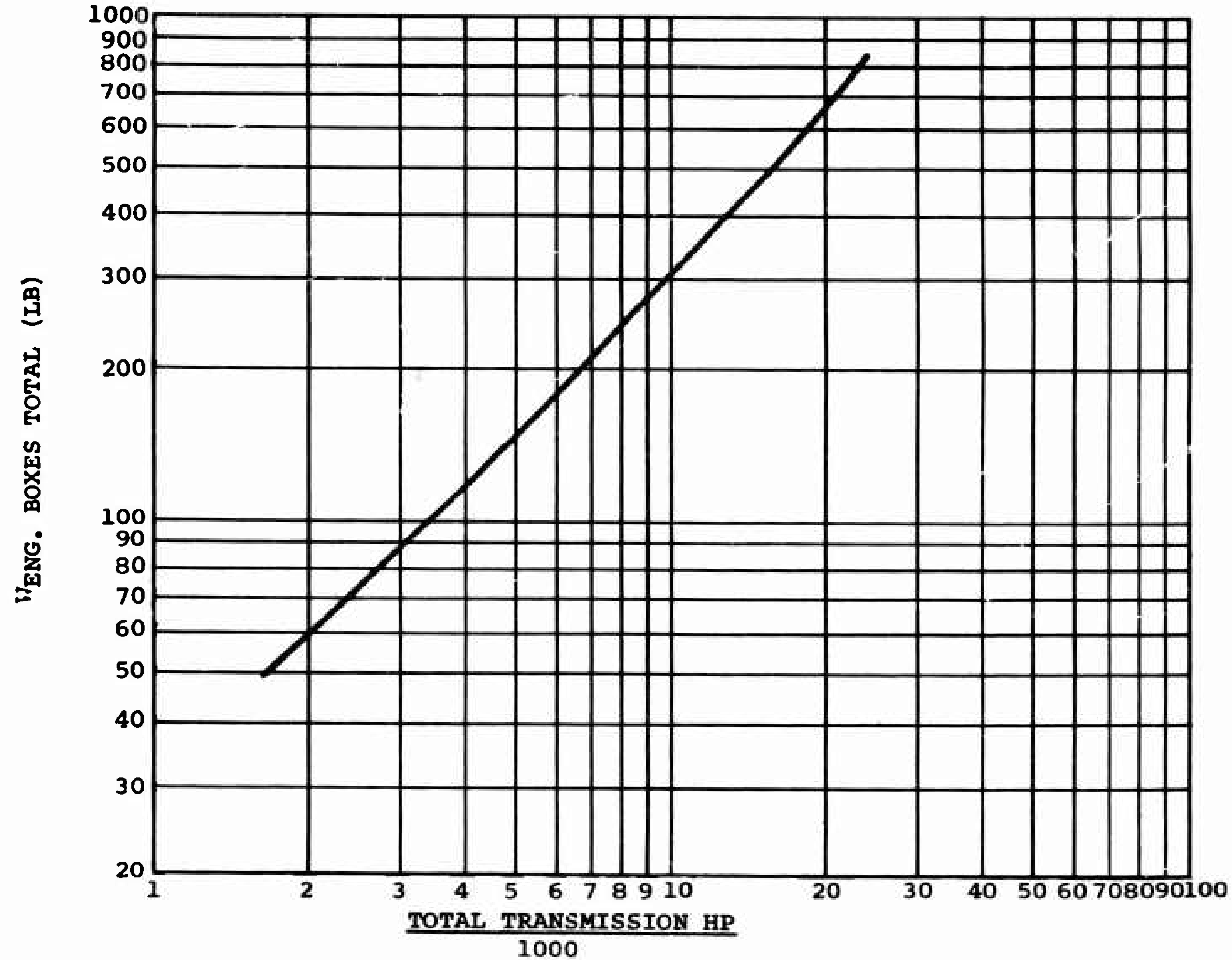


Figure 74. Tandem and Single Rotor Engine Box Weight for System 3

were closely grouped. A major factor restricting location of components from the upper midsection of the fuselage or limiting the possibility of running gas ducts through this area was the clearance requirement due to the forward rotor maximum blade droop angle over the fuselage. In the case of winged aircraft, this restriction was increased because of potential interference problems between the wing-tilt carry-over structure and mechanism and the propulsion components. Thus, a propulsion system arrangement having some gas-coupled components located forward and some aft on the upper fuselage appeared to be a poor design approach.

Investigation of the possibility of locating all propulsion components in the upper forward section of the fuselage near or around the forward rotor pylon was also conducted; but space limitations were severe, with the common power turbine being the major problem. The turbine could not be located forward and yet have adequate clearance for continuous local structural framing around the payload compartment beneath and for rotor blades at maximum droop angle above. Designing for the required clearance would substantially increase the fuselage frontal area. Another undesirable condition that would result from locating the fans forward of the wing was fan hot gas efflux impingement directly on the wing in cruising flight. It was believed that both gas generators and fans should be located high on the fuselage to limit foreign object damage during operations near the ground. The wing was also located in a high position to effect clearance for tilting. In addition, it was felt that a forward location of propulsion items would present undesirable sources of noise adjacent to the cockpit area.

In the case of composite tandem aircraft, the requirement of providing space for stowage of the folded forward rotor blades practically eliminated the possibility of placing propulsion items in a forward position, since all items would have to be located outside the contours of the fuselage and blade stowage bay.

A major reason for investigating forward or midship locations of gas-coupled propulsion components was the concern for longitudinal balance of the aircraft if aft locations were selected. A tendency for too-far-aft location of the center of gravity was of particular concern on tandem composite aircraft, since the weight of a rear-rotor fairing was added and since the blade fold and stow aft in cruise flight. In this study, however,

the engine manufacturers' weights for advanced state-of-the-art propulsion items are considerably lighter than those of current equipment. This is an alleviating factor when considering the aft location of powerplant items.

Selection of a rear location for gas-coupled propulsion components in a high-mounted closely coupled arrangement was, therefore, made for all types of tandem rotor aircraft; and satisfactory balance resulted. The location and component arrangement for system 1a in a lift/propulsion compound configuration is shown in Figure 111; it is typical for the cases of propulsion-unloaded compound and composite tandem rotor types as well.

The common power turbine assembly was located low within the base of a large rear rotor pylon near the point of maximum thickness. The pylon chord was made large to achieve an acceptable thickness ratio. The power turbine assembly was mounted sufficiently low to center around the synchronizing shaft so that a separate turbine drive system into the rear rotor gearbox would not be required. Extremely careful structural design of the pylon would be necessary to ensure adequate accessibility for maintenance and removability characteristics of the power turbine assembly. Cooling could also be a problem.

The power-turbine synchronizing shaft-speed reduction gearbox was mounted on the forward face of the turbine assembly with the output shaft running concentric. The exhaust system from the turbine unit is of the bifurcated type to allow passage of gas outward around the rear-rotor-drive gearbox assembly. The gas generators and diverter valves were located forward of and above the power turbine so that the gas entry points to the turbine scroll section were near the top. This arrangement permitted gas generators and valves to be mounted externally to the main structure in a manner providing easy accessibility under the nonstructural cowling. Gas generator nacelles were faired into the rear pylon lines. The fan assemblies were located directly adjacent to the diverter valves to permit the shortest possible gas duct length. The ducts connecting valves and power turbine were essentially single elbows, thus achieving the shortest possible duct length in this direction of flow. The vertical position of the fan assemblies can be varied to some extent in this arrangement, depending on the results of tests to determine optimum location with respect to inlet flow patterns generated by rotor downwash and wing wake in various flight conditions.

In the composite aircraft type, the masking of the gas generator inlets by the forward rotor fairing could cause serious problems. Selection of the aft location of gas-coupled propulsion components made it necessary in some scaling cases to increase the overall fuselage lengths slightly with respect to the basic length used with other propulsion systems. Scaling of length was done as a function of gas generator airflow as noted earlier. In cases of the larger rotor-diameters and higher design airflows, an increase was required because the gas-coupled items were sandwiched longitudinally between the droop clearance envelope of the forward rotor blades and the rear-rotor-drive gearbox.

Although not shown in Figure 111, aircraft accessory drive system components would be located in a bay between the two gas generators forward of and below the power turbine assembly, or, if detailed design layout indicated a difficult accessibility problem, these components would be moved to a position aft of the rear rotor transmission.

Rotor synchronizing shafts of all the tandem aircraft run forward from the power turbine assembly above the fuselage frames and bulkheads for easy accessibility. In the case of winged aircraft, the rotor synchronizing shafts run above the carry-through structure of the wing and wing tilting mechanism to allow a continuous wing support across the fuselage.

Analysis indicated that proper balance of all aircraft could be maintained with propulsion systems located and configured in the manner described. The slight additional length required between rotors in some scaling cases using a gas-coupled system would increase rotor center distance and move the rotor lift center aft so that its relationship with the estimated center of gravity for these cases would be within allowable limits. In winged aircraft types, the wings would be positioned longitudinally so that the cruise flight center-of-gravity range would be within acceptable limits in terms of percent of mean geometric chord.

Single Rotor Gas-Coupled Systems 1a and 1b

Longitudinal balance and foreign object damage considerations dictated that the gas-coupled propulsion components of single rotor aircraft be mounted forward and high on the fuselage, with the external contours of all items except cruise fans being blended into the main rotor pylon. This location allowed a short-

coupled arrangement of gas-connected parts to be employed in a manner similar to that used in tandem rotor aircraft with the same attributes of minimum power loss, weight, and vulnerability. Forward location of these propulsion elements also matches well with the choice of a low wing arrangement on the single rotor winged aircraft types, since the wing would then be well clear of the efflux from both fans and power turbine, and no hot gas impingement problem would exist in any flight mode.

The propulsion systems were located on the lift/propulsion-unloaded compound aircraft, as shown in Figure 113. The arrangement is also typical for cases of propulsion-unloaded and composite single rotor aircraft types.

The power turbine assembly, as the largest single propulsion item, was located in the base of the main rotor pylon just forward of the rotor-drive gearbox in such a manner that the turbine exhaust could pass to one side of the gearbox and exit approximately at the rotor centerline station. The input drive shaft from the turbine reduction gearbox located at the forward end of the power turbine ran concentrically within the turbine unit to the rotor gearbox. Further detailed layout design might show the desirability of integrating the two gearboxes in a common housing; however, an investigation of this nature was beyond the scope of the present study. The power turbine assembly was mounted above the fuselage bulkheads and frames surrounding the payload compartment; however, integration of this unit with the prime pylon structure would have to be carried out very carefully in detailed design to ensure adequate accessibility for maintenance and removal.

In all single rotor aircraft types employing the gas-coupled propulsion systems, the gas generators and diverter valves were forward of the power turbine, and the fan units were mounted on short pylons directly beside the diverter valves. Vertical position of the fans could be varied to some extent, and final location would be based on results of a further study program. In the case of the compound aircraft, only short elbow ducts were required between valves and turbine. In the composite types, however, an additional short length of duct was required because of a necessity to move gas generators downward and forward to effect clearance with the main rotor fairing in its retracted low-speed flight position. Because of this requirement, the ducting entered the turbine unit scroll from a lower vertical position on composite aircraft. Closer gas generator centerlines were then allowed.

While no major problems of clearance were encountered between propulsion and airframe items on the compound aircraft, design of a single rotor composite machine was made more difficult because of potential interferences between the retracted main rotor fairing and the propulsion units. The main rotor pylon height was greater on the composite-type aircraft. This is because of the vertical distance required between the cruise and hover position of the rotor fairing with rotor blade droop angle considered and because of the lower limitation on the retracted blade fairing imposed by the gas generator cowlings and adjacent parts. To minimize this lower limitation on retracted fairing location, the gas generators were moved forward and were lowered to the maximum extent possible commensurate with good fuselage structural arrangement.

The tail rotor-drive system was conventional and posed no special installation problems in the compound aircraft. In the composite aircraft type, the tail rotor would be stopped and locked in a fixed position in the high-speed flight regime.

No detailed study was made of the location of aircraft accessory drive components; however, initial planning indicated that this system should be located between the gas generators and forward of the power turbine on compound aircraft. The closer spacing of gas generators on composite-type aircraft might dictate positioning some or all of the accessory drive system aft of the main rotor gearbox.

Tandem Rotor Shaft-Coupled Systems 2a and 2b

In each of these systems, two steps were employed in the design integration studies. One was to determine the optimum relative groupings of major propulsion system items, and the other to select the best location of these configurations on the aircraft. For both remote 2a and concentric convertible 2b types of systems, it was desired to employ short shaft lengths between components and to use the least number and simplest type of gearboxes to minimize weight and complexity. In each case, a standard relative grouping was selected for gas generators, cross-shafting, fans, and the gearboxes coupling power to the rotor transmission.

In the remote gas generator system, the power turbine output shafts were as short as clearance of the exhausts from other propulsion components would permit. The gas generator exhaust pipes exited on the skin line at an approximate 45-degree angle

to avoid interference with the engine gearbox, cross-shafting, and fans. With the relationship of cross-shaft and gas generators thus determined, the cruise fans were positioned on the cross-shafts in a manner so that they cleared the exhausts and permitted use of a simple right-angle bevel gearbox outboard and just aft of the fans. The cross-shafts formed a shallow upright vee in an end view, with the apex at the transmission synchronizing shaft and with the gas generators and fans positioned on the legs of the vee. The coupling gearbox on the synchronizing shaft was of the spiral bevel gear type to accommodate this cross-shaft arrangement. The vertical locations of gas generators and fan assemblies were related by their respective lateral offsets from the aircraft centerline. The straight legs of the cross-shaft vee permitted use of a single bevel gear set in the engine gearboxes, and layout investigations indicated that this arrangement permits sufficient design flexibility for location of both engines and fans.

In the convertible system employing gas generators integral with the fan assemblies, the coupling gearbox on the synchronizing shaft is similar to the other shaft system while the included angle of the cross-shaft vee is dependent on the relative vertical locations of the synchronizing shaft and the convertible fan assemblies.

In the case of either shaft drive system, it was theoretically possible to locate the selected grouping of propulsion components anywhere along the synchronizing shaft from the forward rotor transmission to the rear rotor gearbox. Consideration of a location near the forward rotor pylon created problems of interference with forward rotor blades at maximum droop angles, with hot gas impingement on the high wing employed in winged aircraft types, and with a possible high noise level in the cockpit.

In the case of tandem rotor composite aircraft, the requirements of a stowage bay for the folded blades of the forward rotor further mitigated against location of propulsion components in a forward position. A midship location of propulsion items was ruled out because of blade droop clearance requirements on the forward rotor and because of interference with the wing carry-through structure on the winged aircraft. Balance checks of the tandem rotor aircraft indicated that an aft location of propulsion systems was feasible. Therefore, this location was selected for both shaft drive propulsion systems on all tandem aircraft with components placed between

the forward rotor blade droop clearance envelope over the fuselage and the rear rotor drive transmission. The arrangement of these two shaft drive systems on the tandem rotor lift/propulsion-unloaded compound aircraft is shown in Figures 116 and 120.

The design integration of shaft-coupled propulsion systems did not require extension of the basic fuselage length in any aircraft type over the scaling range of interest. Whether the coupling gearbox employed in these shaft propulsion systems could be integrated with the gearcase of the aft rotor transmission would depend on detailed design layout and aircraft balance investigations beyond the scope of this study. It has been assumed that the two gearboxes are separate units.

In the case of both shaft-driven systems, the coupling gearbox was located on the rotor-drive synchronizing shaft just forward of the rear rotor transmission and low within the aft pylon. The vertical location of gearbox and synchronizing shaft and the resulting cross-shaft dihedral angles are a function of the type of aircraft concerned. In the propulsion-unloaded type, the absence of a wing and associated carry-over structure allowed vertical location of the synchronizing shaft in a low position just above the structural framing around the payload compartment. Without a wing, the cruise fans could be located in an optimum vertical position from a thrust pitching moment point of view, without concern for wing wake effects on fan inlets. However, choice of too low a fan position would also lower the engines in the remote gas generator system and complicate gas generator installation problems, including inlet design. In the convertible system, no such problem arises.

Presence of a high wing on the lift/propulsion-unloaded compound aircraft required raising the synchronizing shaft and coupling gearbox above the wing section, since it was not considered to be advisable to pass the shaft directly through the wing carryover structure, particularly in cases where the wing was tilted, and a tilting system was required in the same area.

The installation aspects of both shaft-coupled systems in tandem composite aircraft were generally similar to those involved with lift/propulsion-unloaded compound types. Since the propulsion components were located aft in both cases, no component interference problems were encountered with stowage requirements of the forward rotor system. The synchronizing shaft in the rotor-drive system was vertically located below the forward

rotor stowage bay and above fuselage framing and wing carryover structure. Studies indicated a potential problem of interference between cruise fans and the rear-rotor fairing in its retracted (hover) position; however, this problem could be avoided by a small increase in rear-rotor pylon height. On the composite types the large forward rotor fairing body tends to mask the inlets of the gas generator in the remote shaft-driven system (2a) which could create a serious inlet flow problem.

Investigations of the two shaft-coupled systems indicated a clear superiority for the convertible system over the remote type from all design points of view. These include relative simplicity in terms of numbers of major components, ease of design integration with airframe, flexibility in terms of vertical location of fans, and maintainability aspects of the system.

Single Rotor Shaft-Coupled Systems 2a and 2b

Design investigations of the two shaft-coupled systems in single rotor aircraft involved the same procedure of determining optimum relative component grouping and selection of the best location for these groupings on the aircraft. The selected relative groupings of gas generators, gearing and shafting, and fans is the same in these single rotor cases as in the tandem cases discussed previously, and for the same reasons. Location of the propulsion elements forward and high on the aircraft was selected on the same basis as was indicated earlier for single rotor gas-coupled systems, the predominating criteria being proper aircraft longitudinal balance and minimum foreign object damage possibilities. The integration of the systems with the single rotor lift/propulsion-unloaded compound aircraft types is shown in Figures 118 and 122.

In the remote shaft-driven system 2a, it was necessary for balance purposes to locate the gas generators and fans far forward. This placement required that the cross-shaft and coupling gearbox of the system be located well forward of the main rotor gearbox, with a longitudinal main drive shaft connecting the two. A review of alternative configurations with cross-shafts driving directly into the main rotor gearbox and gas generators in a forward position was made. These configurations were rejected since the length of the high-speed engine output shaft was unduly extended and the position of the cruise fans was such that ingestion of exhaust gases appeared likely.

The forward location of the cross-shaft in the selected configuration placed the fans in a position sufficiently forward with respect to engines to minimize the possibility of exhaust ingestion. The fans were also located sufficiently above the engines by means of a vee or dihedral cross-shaft arrangement and by use of a downward sloping longitudinal drive shaft from the rotor to the coupling gearboxes to allow the engine exhaust to exit below the fan inlets.

The forward location of gas generators on this remote fan drive system was constrained in terms of vertical clearances by the structural requirement for continuous fuselage framing beneath the engine bay on one hand and the droop clearance envelope of main rotor blades above the engine cowling on the other.

The design investigations of single rotor composite aircraft employing the remote shaft-driven system made apparent an additional problem of potential interference between engine cowlings, fans, and the retracted (during low-speed flight) main rotor fairing. When the rotor fairing is lowered sufficiently to clear the blade droop envelope, an interference may occur, depending on aircraft scaling, unless the location of cruise fans is as required to give clearance from engine exhaust impingement. The only solutions to this problem are to increase rotor pylon height to provide clearance, to increase the lateral offset of the fans, or both.

In the case of the single rotor composite aircraft with convertible fan system 2b, the fact that the engines and fans were concentric removed the engine exhaust fan location problem. In addition, the convertible system fan assemblies could be moved forward, though at the expense of a longer longitudinal drive shaft. The fans could also be lowered to the point where interference possibilities with a retracted main rotor fairing would be eliminated.

Of the two shaft-driven systems, the convertible fan type was distinctly preferable from all installation design viewpoints, as was true in the case of tandem rotor aircraft.

Tandem Rotor Independent System 3

Initial design investigations were made of the independent propulsion system 3 in tandem rotor aircraft on the assumption that the fans would be employed only in cruising flight. The significant difference between this propulsion system and

integrated systems with respect to design installation was in the freedom to separate the cruise fans from the lift system propulsion components. No cross-shafting or ducting to the fan was involved.

For reasons outlined previously in other tandem rotor cases, a selection was made of an aft location for the lift system shaft turbines and the gearboxes coupling power to the synchronizing shaft. A review of potential locations for the separate cruise fans was conducted. Forward locations on the fuselage were eliminated because of hot gas impingement problems on the wings of the lift/propulsion-unloaded compound and composite types in cruise flight. In addition, it was considered that a possible high noise problem in the cockpit area should be avoided. In the case of a propulsion-unloaded aircraft, the forward fan location was also rejected, primarily because of possible gas reingestion in the lifting system engines.

Considerable attention was devoted to the possibility of locating the fans in a typical under-wing nacelle location on the winged types. It was necessary to consider both the cases of a tilting and a nontilting wing, since scaling of the aircraft and resulting tandem rotor disc spacing determined whether or not the wing would be tilted. If it was tilted, a standard swept-forward pylon mounting of fans resulted in interference between fans and the blade droop clearance envelope of the forward rotor with the wing in the up position. If the wing was not tilted, a standard pylon mount for the fans resulted in a fan inlet height above the ground (or water, if a water-adaptable version was considered), low enough to give concern over fan ingestion problems during engine idling conditions, particularly under combat conditions.

It was assumed that the cruise engines of the independent system would be operated at idle power during vertical operation as a standard procedure in the case of no-fan-lift augmentation. A further consideration in the under-wing fan was the spanwise location of fan assemblies on the wing. Review of positions close to the fuselage indicated a probable high interference drag characteristic with fuselage and landing gear sponsons. A location substantially farther outboard on the wing gave concern for engine-out thrust unsymmetry problems in high-speed cruising flight.

An equivalent position of fans considered for propulsion-unloaded tandem rotor aircraft was a pylon mount from the

fuselage longitudinal midsection. If the nontilting fans were mounted high, the lift system engine reingestion problem reappeared. If mounted at a midfuselage waterline or below, the foreign object damage situation with idling engines was again of concern. In this latter case, a difficult structural mounting problem for the fan assemblies was also present.

The selected location for the fan assemblies on all tandem aircraft types, where fans provide no hover thrust, was high and to the rear on the aircraft fuselage adjacent to the shaft turbines of the lifting system. The configuration is thus generally similar to the integrated remote shaft-coupled system 2a except for the absence of cross-shafting to the lift system (see Figure 124). Since no fan cross-shafting was involved, the cruise fans could be moved forward to some degree with respect to the lift system engines; they could also be placed in a position where shaft turbine exhaust would not impinge on or be ingested by the fans. However, the selection does create difficult combined problems of lift powerplant installation and fan structural mounting since four engines in close proximity are involved.

The fan pylon mounting structure required could interfere with the accessibility and removability characteristics desired for the lift system shaft turbines. Layout design in sufficient detail to assess all the problems that might be encountered in this area was beyond the scope of the study; however, it was clear that this independent system posed many more design installation problems than the integrated systems studied. One of these was a tendency for the aircraft center of gravity to move further aft than desired.

<u>Single Rotor Independent System 3</u>

Design study of the independent propulsion system installation on single rotor aircraft types, assuming no use of fans as a vertical lifting aid, was conducted based on the same general considerations employed in assessing other configurations. The lift system shaft turbines were located high and forward on the aircraft to minimize foreign object damage and maintain aircraft balance. The location of cruise fans with respect to the shaft turbines was again primarily a function of avoiding engine exhaust impingement on the fans and exhaust ingestion. Therefore, the fan assemblies had to be forward of and separated vertically from the exhaust sections of the shaft turbines. Since it was impossible to conduct detail balance

calculations for all airframe-propulsion design integration studies, the exact location required for all items was not determined, and the major variable item with balance would be the length of the longitudinal drive shaft to the main rotor gearbox.

The selected propulsion configuration for this case is shown in Figure 125. The design installation complexity of locating four engines together in the same area improved somewhat over the tandem rotor case, since the fans could be mounted directly off the upper shoulders of the fuselage in a main longeron area, and since they should not interfere with mounting of and accessibility to the bay for the shaft turbines. Since a low wing was employed in the single rotor aircraft, the exhausts of all engines could be arranged to clear airframe components. However, local fuselage protection against high exhaust temperatures must still be provided. Detailed balance calculations might show that a danger to personnel from engine exhaust near the forward door of the payload compartment existed, depending on the exact relationship of exhaust and door. Since the vertical location of fans was independent of shaft turbine location, no fan clearance problem with the retracted rotor fairing of the composite-type aircraft was present. In aircraft types where the flight regime requires combined operation of all engines, the cockpit noise would undoubtedly be high.

As in the tandem rotor cases involving independent systems with no fan lift assist, there are several potentially serious propulsion system design installation problems. These problems are mainly a function of the number of separate major components required in the system.

Fan-Augmented Vertical Lift

Discussion thus far of independent propulsion system 3 has not covered the case of fan tilting or fan exhaust deflection to supplement rotor lift during vertical operation. It appears advisable to consider these possibilities for reducing the total rotor thrust and installed power requirements during hover flight. The fans in this propulsive configuration would be installed separately, and would be available for lifting use if their thrust could be properly directed. Even if not tilted or deflected, they would probably be run at idle power as a standard procedure during vertical operation, thus consuming fuel in any case.

In addition to aspects of airframe design integration feasibility of tilting/deflecting fans, certain basic operational problems would result from their use:

1. Fan Downwash Velocity - The downwash velocity of highly (relative to rotors) disc loaded fans is on the order of 400 to 800 feet per second, depending on bypass ratio. It is doubtful that downwash velocities of this order would be tolerable in an Army forward-area environment, since problems of recirculation, foreign object damage, heavy dust signatures, poor flight crew visibility, and hazards to ground personnel and equipment could result. If a requirement for normal operation from water areas is also to be considered, the effect of these downwash velocities as a disturbing factor on the water and on the aircraft could be a problem.

2. Fan Exhaust Gas Temperature - If engine exhaust is diverted downward, hot gas plumes of approximately 1000°F impinge on the ground near the aircraft, posing potential hazards to adjacent ground personnel and equipment. Some improvement could be obtained by attempting to mix the cooler fan efflux or ambient air with the engine exhaust, though a device for this purpose should not unduly penalize the system in terms of additional weight or cruise flight drag.

3. Fan Noise Signature - A substantial increase in aircraft noise signature during vertical flight could result from full power operation of the fans as opposed to idle operation.

It would thus appear that employment of these cruise components to aid the lifting rotor requires operational acceptance of problems commonly attributed to much higher disc loaded VTOL concepts in an aircraft whose basic concepts are set up to avoid these problems.

From a vehicle safety standpoint, the major problem is the consequence of a fan engine failure in hover flight. A sudden loss of all or a major part of the power on one side of the aircraft could result in a vehicle's rolling acceleration greater than that available from rotor control counteraction.

In such cases a particularly swift and automatic sensing of the disturbing acceleration would be required so that the thrust of the remaining fan could be lowered or cut completely. If the fans are initially providing a substantial percentage of the hover lift, vertical equilibrium would be greatly affected and a sudden increase in rotor lift would be required. If fast-acting automatic controls are employed for such an emergency, the control sensitivity and authority would have to be worked out carefully to avoid a possibility of inadvertent action during normal pilot-induced maneuvers.

Another way to protect against a large amount of asymmetric lift with a fan engine failure is to provide an intershafting or interducting system between the two fan engines so that either or both can power both fans. This means the addition of weight and complication because of the cross-tie arrangement, power pickoffs from the engines, and a means for eliminating rotation of a disabled engine.

In the case of a tilting fan arrangement, the proximity of the plane of the rotor disc can have a significant effect upon the magnitude of rotor vibratory forces, depending on the exact geometry of the particular design. With tandem rotor configurations, the existence of these high-pressure sinks directly under the rotor blades can cause a 2/rev/blade vibration in both rotors estimated as more than 0.04g.

Certain additional design installation factors are important when considering the feasibility of tilting or deflecting cruise fan thrust during hover flight. Among these are the following:

1. Fan location to avoid any large longitudinal pitching moments from fan thrust throughout the tilting or deflection angle range, or large changes in fan static weight moment when tilting.

2. Avoidance of fan exhaust impingement on parts of the airframe.

3. Requirement for design of a high-reliability synchronized fan tilting system.

4. Design of efficient thrust deflecting devices which minimize exhaust system losses in cruising flight.

The following paragraphs discuss the general results of a study of design installation feasibility of cruise fan thrust vectoring on the various aircraft types from the airframe integration point of view.

Tandem Rotor Aircraft - Design investigations of propulsion-unloaded aircraft showed that the absence of a wing permitted the possibility of mounting tilting fan systems on the sides of the fuselage in a location approximately coincident with the aircraft center of gravity, thus eliminating or minimizing the requirement for trimming fan thrust moments through the range of tilt angles. However, the required location of the tilting axis would make the structural mounting and tilt system design very difficult.

Even if the fan inlet were located just below the maximum droop clearance envelope of the forward rotor blades, the tilt axis of the fan assembly would lie below the fuselage level corresponding to the top of the payload compartment. The vertical distance would depend on the fan size in a particular case.

With a fuselage frame radial thickness of about 4 to 5 inches, a mounting incorporating a synchronized tilting mechanism would be structurally inefficient if not totally impractical. If it became desirable to install an intershaft system between fans to preserve symmetrical engine-out thrust, the cross shaft would have to run directly across the payload space, an obviously unacceptable situation. In addition, use of an average fan/engine size showed that the gas generator tail-pipes would be located approximately 4 to 5 feet above the ground. It was considered that this location would cause serious effects on the ground and would not be acceptable for water operation since the hot tailpipes would almost touch the water at a zero roll angle. In addition, the tail-pipe location was judged to be too close to both sponson fuel tanks and main landing gear, although the mounting pylon could be lengthened laterally to alleviate this condition. This tilting installation was not considered to be feasible from a total design installation viewpoint.

The general installation characteristics of a deflected thrust arrangement were also reviewed for applicability to a propul-sion-unloaded tandem rotor aircraft. In this case the cruise fans would be fixed on a pylon off the upper fuselage shoulder with the structural support beam running across the upper

longerons and beneath the transmission synchronizing shaft. The fans would be located longitudinally on the fuselage so that the net line of action of the gases would run nearly through the center of gravity of the aircraft at all deflection angles, and the fan contours would clear the maximum droop envelope of the rotor blades.

Two potential problems were considered which, it is believed, could be overcome satisfactorily. The first was the heating effect of deflected exhaust on fuel sponsons and landing gear. The span of the mounting pylon could be increased to alleviate this problem and the exhaust thrustline canted several degrees laterally outward until no impingement occured. If the cant angle were small, very little vertical fan thrust would be lost in the system. An alternative would be to use a deflection device turning only the fan bypass air, and allowing the engine exhaust to pass straight aft at all times; however, it is not known whether such a device could be configured in an efficient and practical manner.

The second concern was the possibility of ingestion of fan system exhaust into the inlets of the rotor-drive-shaft turbines in cruising flight, since fans are mounted well forward of the lift system engines. However, the relative lateral and vertical displacements of fans and shaft turbines together with the effect of rotor wash on fan efflux would appear to lessen the possibility of reingestion. In general, the design configuration was considered to be practical, although a detailed flow investigation program was believed to be necessary.

Review of the possibilities of locating tilting cruise fans on lift/propulsion-unloaded tandem rotor winged compound aircraft indicated that a pylon mounting off the wing underside could be effected whether the wing tilted or not. In the case of wing tilt, the fans would be attached rigidly to the wing at inboard locations. If the wing were fixed to the fuselage, the fans could be trunnion-mounted from attachments ahead of the forward spar and tilted independently of the wing. In either case, the fan nacelle would have to be located sufficiently aft on the wing chordwise so that the fan front face would clear the rotor droop clearance envelope. This position leaves the engine exhaust very close to the ground when the fans are tilted up.

Location of fans on the wing minimized the longitudinal pitching moment arm from thrust centerline to aircraft center of

gravity in the hover position. However, this distance would still be about 3 to 4 feet and would involve the requirement for a significant amount of trim by differential collective rotor blade angle. Rearward movement of the wing to reduce fan moment arm to zero would upset the desired wing center-of-pressure and aircraft center-of-gravity relationships.

Spanwise location of fan pylons on the wing should be as close inboard as possible (without incurring exhaust impingement on the airframe or high interference drag effects with fuselage and sponsons), so as to minimize the effect of lateral thrust moments in an engine-out condition. This arrangement was considered to be practical.

Use of a deflected thrust arrangement for the cruise fans in this aircraft type was considered for both tilting and non-tilting wing cases. If the wing was tilted, it seemed a poor choice to mount the fans independent of the wing and then to deflect fan exhaust, since the wing was a ready mount for tilting. In addition, a fuselage mounting at the midsection area is structurally difficult, the fan assemblies must be mounted much too low on the aircraft from a foreign object damage viewpoint, and the fan exhaust would impinge on the lower wing surface in cruise flight. The arrangement was thus considered to be impractical.

If the wing did not tilt, the fan assembly could be pylon-mounted beneath the wing as described above and employ a thrust-vectoring nozzle system. Here the line of action of the deflected thrust could be made to act through the aircraft center of gravity by proper longitudinal positioning of the nacelles on the wing. The main problem associated with this configuration is a low fan nacelle inlet even when the nacelle is located as close to the wing as possible, thus providing a high susceptibility to foreign object damage in a rough-field operating environment. This problem was considered to be sufficiently serious to label the combination impractical.

Design integration reviews of tandem rotor composite aircraft with independent fan thrust vectoring yielded results similar to those noted previously for tandem rotor winged compound aircraft. A further consideration in composite cases was the possible clearance problem of tilting fan nacelles with forward rotor bay fairing doors in the vertical flight configuration. Tilting fans might have to be moved further out spanwise on the wing to correct such a problem.

Single Rotor Aircraft Types - On propulsion-unloaded aircraft, the requirement for location of tilting fans at or near the aircraft longitudinal center of gravity to minimize longitudinal trim problems through the tilting range indicates a placement of the fan tilt axis directly beneath the center of the main rotor. This axis can be sufficiently high on the aircraft to allow a good structural pylon mounting and to provide for a proper interconnected tilting system design configuration. However, a possible space interference problem with the main rotor transmission is indicated. This location gives an exhaust ground clearance in the hover position of about 8 feet, a distinct improvement over the equivalent tandem rotor case.

A similar problem of exhaust impingement on the sponsons housing fuel and the main landing gear is present, however, unless the fan pylons are lengthened sufficiently to allow exhaust clearance from airframe. Greater lateral offsets result in proportionate increases in fan asymmetric thrust under engine failure conditions for the hover and cruising flight modes. The major problems expected from this configuration, however, would be the possibility of lift system shaft turbine exhaust ingestion by the fans in the cruise-flight mode, with exhaust impingement on the fans also probable. On this basis, the design configuration was rejected.

A thrust deflection system on this aircraft type suffers from the same p oblem of exhaust impingement on sponsons in the vertical lift position unless fan lateral offsets are fairly large and/or the exhausts are canted outboard. It appears possible in this case, however, to locate the fans sufficiently forward, with the exhaust line of action still near the center of gravity to eliminate the fan ingestion problem. Impingement of shaft turbine exhaust on the fans still appears to be probable unless long shaft turbine tailpipes are employed. This configuration appeared to be marginally acceptable.

Review of the possibilities of installing a tilting fan system on the lift/propulsion-unloaded aircraft type showed that the presence of a low wing beneath the aircraft center of gravity made this configuration unacceptable. The fan ingestion problem was also present; therefore, the layout was not considered to be a practical one.

The deflected fan exhaust case was also rejected because of the presence of the wing below the fan exhaust. Even if the

fans were offset longitudinally from the center of gravity in either direction so as to prevent impingement in the hover position, the problem usually would be present at intermediate fan tilt positions.

Results for the composite aircraft type are the same as for the winged compound case previously discussed.

DESIGN OPTIMIZATION TECHNIQUES

Following is an outline of the optimization methods employed to define parameters required for selection of the optimum aircraft of this study. The first portion describes the aerodynamic work in this area.

In general, a specified disc loading and gross weight, along with drive system efficiencies, defined the installed power required for a hovering capability of 6000 feet at 95°F temperature. Rotor shaft horsepower and auxiliary thrust requirements for cruising flight were also defined for each configuration in the matrix of aircraft investigated. After specification of performance requirements, the configuration characteristics were defined by the installed power or by an aircraft constraint. Constraints and configuration definition are discussed in detail in the following paragraphs.

Ground Rules and Constraints

The three aircraft types under study, propulsion-unloaded, lift/propulsion-unloaded, and composite, in both single and tandem rotor configurations, were defined in accordance with the following ground rules and constraints.

Ground Rules - The basic aerodynamic criteria to which all aircraft were made to comply are labeled ground rules. They specify operating conditions and provide a common basis for comparison. The following ground rules apply:

1. Rotors of the propulsion and lift/propulsion-unloaded aircraft were operated at the highest lift at which 95 percent of maximum rotor lift/drag ratio (L/D_E) could be obtained.

2. A wing aspect ratio (AR) of 6 was selected for all lift-unloaded configurations.

3. Wings of the lift/propulsion-unloaded aircraft were operated at a maximum lift/drag ratio (L/D) of 18.45; they were used at an operating lift coefficient (C_L) of 0.54 at the design cruise flight condition.

4. The rotor solidity required for lift/propulsion-unloaded and composite aircraft was defined at the design hover condition of 6000 feet altitude and 95°F temperature by a thrust coefficient (C_T/σ) of 0.11.

5. Composite aircraft transition speed was set at 125 knots, and on this basis a wing loading of 80 pounds per square foot was employed for this type.

6. A sea level maximum speed limit of 350 knots was placed on composite aircraft types based on both a structural dynamic pressure limit commensurate with airframe weight estimates and on a fuselage critical Mach number.

7. Vehicle design cruise speed was defined at a power setting of 90 percent of maximum.

8. All forward flight performance work in the basic study was based on sea level standard ambient conditions.

Constraints - Specific limitations on rotor operation were based on test data and experience accumulated by the Vertol Division of Boeing. Rotors were not operated beyond the following limits:

1. An advance ratio (forward speed/rotor tip speed ratio, μ) of 0.8.

2. An advancing blade tip Mach number.

$$\left(\frac{\text{forward speed + rotor speed}}{\text{speed of sound}} = M_{T_{90}}\right) \text{ of } 0.92$$

3. A plane of no feathering angle (α_{PNF}) not greater than 15 degrees

4. Rotor limitations associated with blade stall as discussed in the section titled Compound/Composite Configuration Studies.

Matching Methods

Application of cruise fans to the propulsion-unloaded, lift/propulsion-unloaded, and composite aircraft appeared to have potential at high speeds where basic rotor system propulsive capability diminished.

The cruise fans used in this study attain propulsion efficiencies equal to or greater than the conventional rotor systems at speeds above 200 knots. The aerodynamic procedure employed in matching cruise fan propulsion systems with each aircraft type is outlined in the following paragraphs.

Propulsion-Unloaded Aircraft - In this case the rotors were operated at 0.95 $(L/D_E)_{max}$ until limited by a 15-degree plane of no feathering. From this point on, the cruise fan systems were required to provide the additional propulsion force while the rotor still provided all the required lift.

In this approach the additional propulsion requirement (cruise fan thrust required) and rotor horsepower were defined for the anticipated range of operating speeds and gross weights.

In the case of integrated propulsion systems, the thrust available (that thrust capability provided by the excess gas generator airflow not used in driving the rotor system) plotted against thrust required defined the maximum and cruise-speed values as shown in Figure 75.

The mission fuel requirement was then calculated by using the 90-percent power setting and pertinent cruise speed in the design mission profile. The values of total required cruise fan airflow, required rotor blade area, and mission fuel were defined across the matrix of study input parameters by repeating the calculations for each combination of disc loading, gross weight, bypass ratio, and type of propulsion system, with the exception of independent propulsion system 3.

In this case, cruise propulsion force was provided by an independent fan system, and maximum speed could not be defined by using excess gas generator airflow. Instead, a desired maximum speed of 270 knots was specified, and the resulting cruise

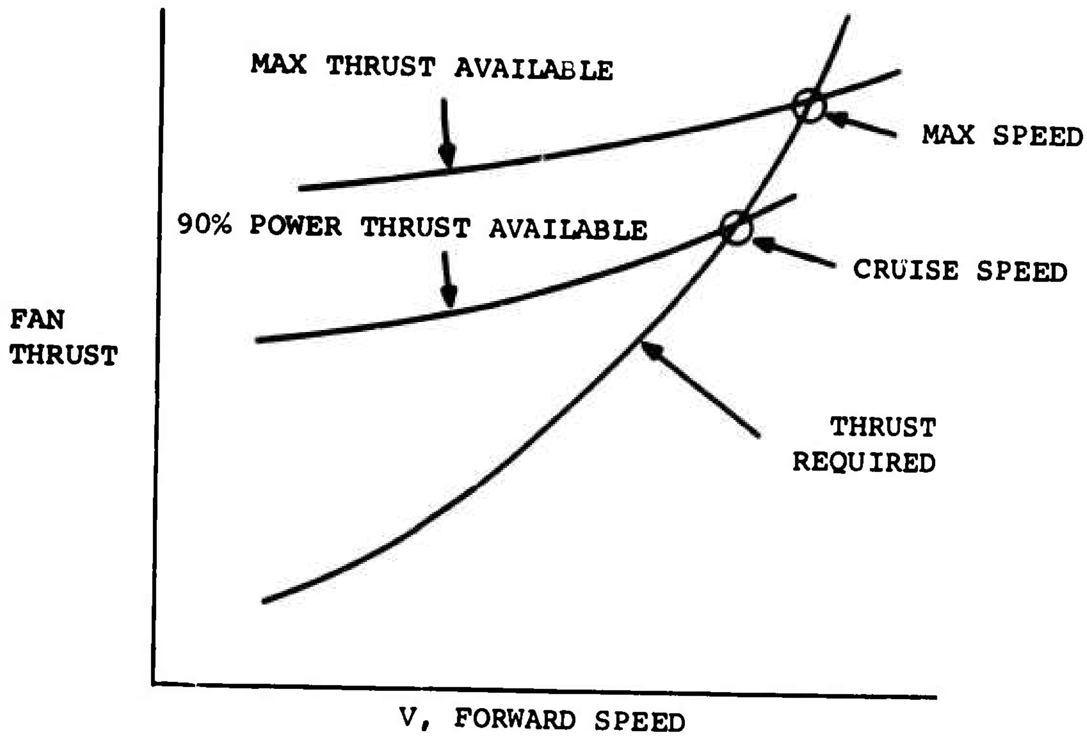


Figure 75. Thrust Available and Thrust Required vs. Forward Speed

fan airflow requirement was determined for a 90-percent power setting and the associated cruise speed. Mission fuel and other required parameters were then computed.

A matrix of candidate aircraft were thus defined in terms of the various disc loadings and bypass ratios of the study for each propulsion system. This procedure enabled selection of an optimum configuration fulfilling all design requirements.

Lift/Propulsion-Unloaded Aircraft - In this type of aircraft a wing was the primary lifting device in cruising flight, carrying approximately 80 percent of the total lift at maximum and cruise speeds. Rotors were operated at a nominal lift value and slowed to provide minimum rotor drag in cruise flight. This minimum rotor drag condition was defined at an advance ratio (μ) of 0.8.

With such rotor operation the cruise fan provided essentially all the propulsive force required to overcome aircraft drag at cruise and maximum speed. As noted earlier, the area of the wing was determined by use of an operating lift coefficient of 0.54 at the design cruise condition. A margin for aircraft maneuver capability was thus allowed.

Required cruise fan thrust values were determined for a range of gross weights for each disc loading. In the case of integrated propulsion systems the thrust available was again computed from the gas generator airflow remaining after deducting the amount required for the rotor to sustain the nominal lift values.

The thrust required and thrust available data again provided the maximum and cruise speeds for the various propulsion systems as shown in Figure 75. The associated parameters required to complete the aircraft design matrix were then computed as indicated previously. Independent propulsion system 3 was again an exception, and a rotor limit speed of 270 knots was utilized in this case to define the comparative requirements of the fan system.

Composite Aircraft - The composite vehicle differs from the other types in that the rotor is stopped and stowed for high-speed flight. In the performance analyses at high speed this aircraft type was treated as a conventional airplane employing cruise fan engines. Transition occurs at 125 knots, and above that speed the cruise fans provide all of the propulsion force and the wings provide all of the lift. The maximum and cruise speed capabilities were again a function of matching the thrust available and thrust required. For integrated propulsion systems the thrust available was provided by utilizing the total gas generator airflow installed for propulsion, this value being based on the installed value determined by the design hover condition as modified by forward flight ram effects. Figure 76 presents a typical example of the available and required thrust variation as a function of forward speed.

Independent propulsion system 3 is an exception to this approach since the thrust available is not defined by the hover gas generator airflow. Independent cruise fan thrust was provided in this case to fix the aircraft maximum sea level speed at the 350-knot limit defined previously.

For these composite aircraft the mission fuel and other data required for the design matrix were computed to complete the performance matching for this type of vehicle.

Mission Definition

A fixed design mission was employed for all parametric aircraft in the matrix. The mission profile, used to define design fuel requirements in all cases, is as follows:

1. Warm-up and takeoff - 2 minutes at normal rated power.
2. Cruise outbound - 100 nautical miles at 90 percent of maximum power.
3. Midpoint landing and takeoff - 2 minutes at normal rated power.
4. Cruise inbound - 100 nautical miles at 90 percent of maximum power.
5. Land with reserve of 10 percent of initial fuel.

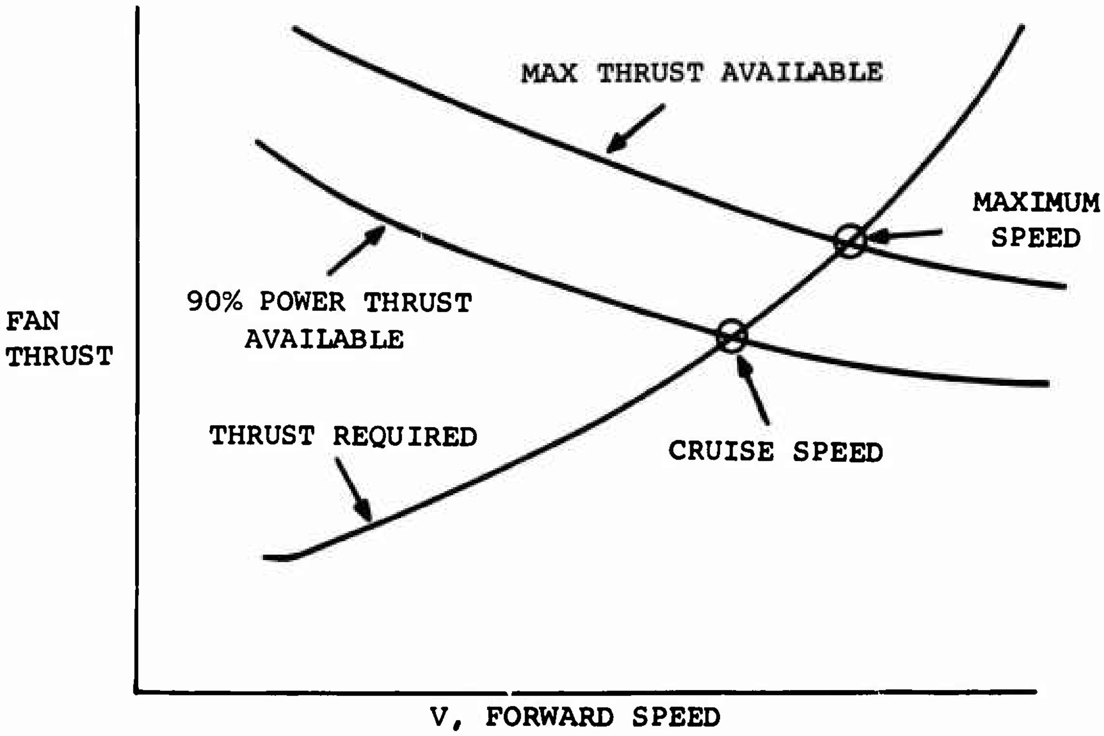


Figure 76. Example of Thrust Available and Thrust Required Variation

For this mission the payload of 6000 pounds was carried both outbound and inbound.

An additional mission was flown to define the ferry range capability of each optimum aircraft. The definition of this mission profile is as follows:

1. Warm-up and takeoff at sea level standard conditions - 2 minutes at normal rated power.

2. Climb on course at the optimum altitude for best range.

3. Land with a reserve of 10 percent of the initial fuel.

For this mission the ferry takeoff gross weight was defined by either a performance limitation or by a reduced flight limit load factor of 2.0, depending on aircraft type.

The performance limitation for the propulsion-unloaded aircraft specified that at the maximum ferry gross weight the minimum power required for level flight, plus an increment for a 500-foot-per-minute rate of climb, be no greater than normal rated power. For the lift/propulsion-unloaded aircraft the same criteria applied. A lift coefficient of 1.5 was used to define the wing lift; this provided the maximum increase in the required gross-weight-to-power ratio.

The composite aircraft ferry mission involved a rolling takeoff similar to that of a fixed-wing airplane. A performance limitation employed for this aircraft type was that the thrust required at the maximum allowable break-ground speed of approximately 125 knots (10 percent above the stall speed) be no greater than the maximum available cruise fan thrust. The maximum airplane lift coefficient employed in this case was 2.4.

Method of Selection of Optimum Aircraft

The study required that 30 optimum aircraft be selected. The combination of aircraft configurations, concepts, and cruise fan propulsion systems making up the 30 aircraft is shown in Figure 77.

The major parameters to be varied to provide a basis for selection of the optimum aircraft in each of the above cases

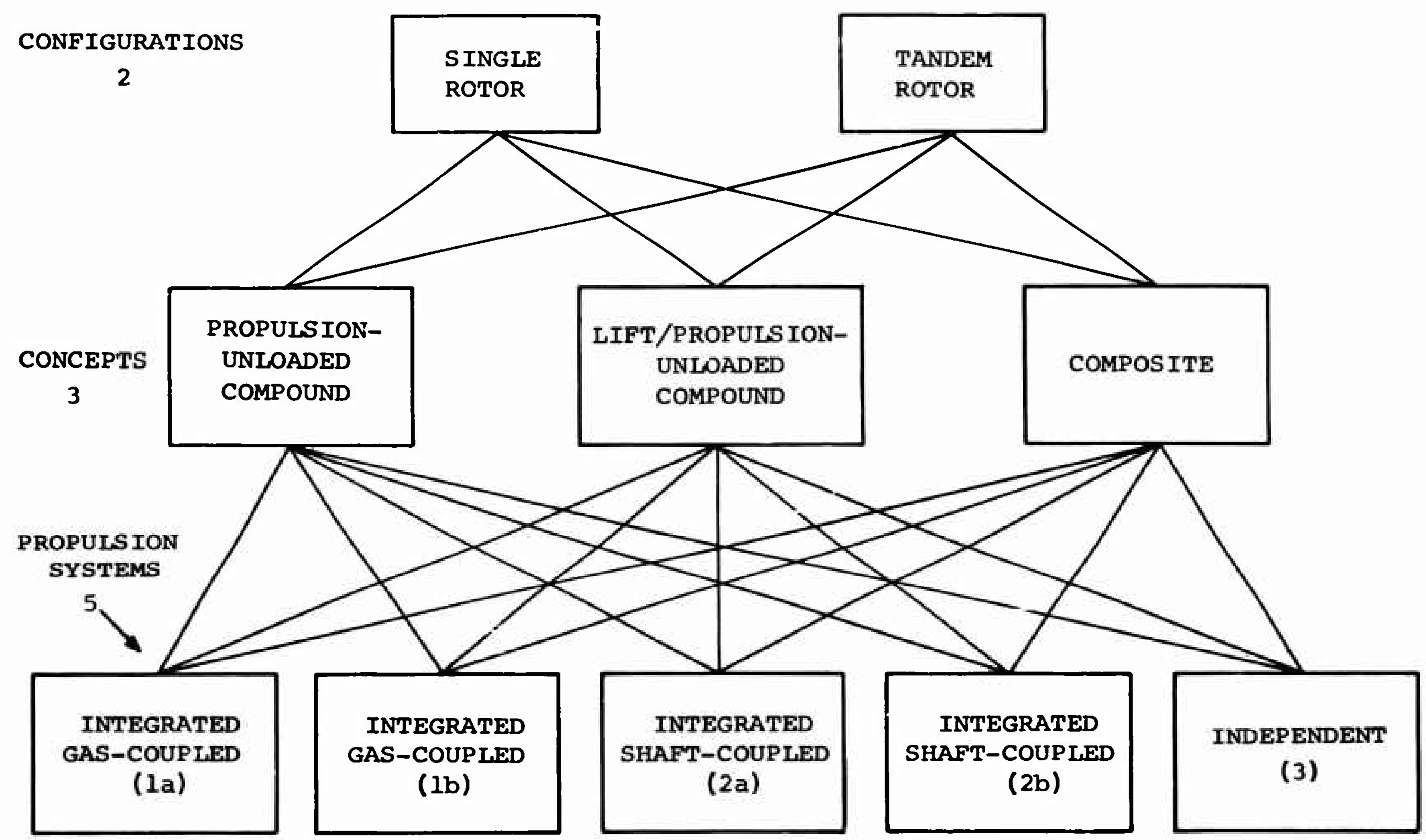


Figure 77. Thirty Selected Optimum Aircraft

are shown in Figure 78. The range of variation is also indicated.

The criterion employed for selection of each of the 30 optimum aircraft was peak relative productivity in the design 100-nautical-mile radius mission profile. Relative productivity was defined as the 6,000-pound design payload times design cruise speed per pound of empty weight.

To determine an optimum aircraft, a unique combination of disc loading, bypass ratio, and gross weight yielding peak productivity had to be defined. Propulsion systems and airframe scaling data in the areas of design sizing, weights, and aerodynamic drag were established over the parameter range of interest. Point design layout investigations sufficient to validate scaling factors were also conducted.

In the case of each optimization, the approach was to analyze the 36 aircraft representing all parameter combinations shown in Figure 78 for the fixed design mission to obtain the required amounts of fuel. In addition, an aircraft group weight analysis was conducted on all combinations to determine available payloads covering the gross weight scanning points shown. This procedure allowed determination of the actual gross weights of 12 candidate aircraft in each optimization, all capable of carrying just the 6000-pound design payload for the mission. Plots of payload required versus payload available, as shown in Figure 79, were used for this purpose.

Empty weights of the 12 candidate aircraft were obtained in each optimization procedure by subtracting from the gross weights the known useful loads, which were composed of the 6000-pound payload, the 750-pound fixed useful load, and the appropriate mission fuel.

The aerodynamic analysis also provided the sea level design cruise and maximum speeds of the candidate aircraft. Cruise speed was defined at 90 percent of maximum power. In the case of aircraft with integrated propulsion systems, the speeds were obtained by matching available and required power in the conventional manner, with available power being determined by the amount installed for the 6000-foot, 95°F design hover condition. In the case of independent propulsion systems, determination of the speeds was based on airframe limiting factors, depending on the aircraft concept being considered.

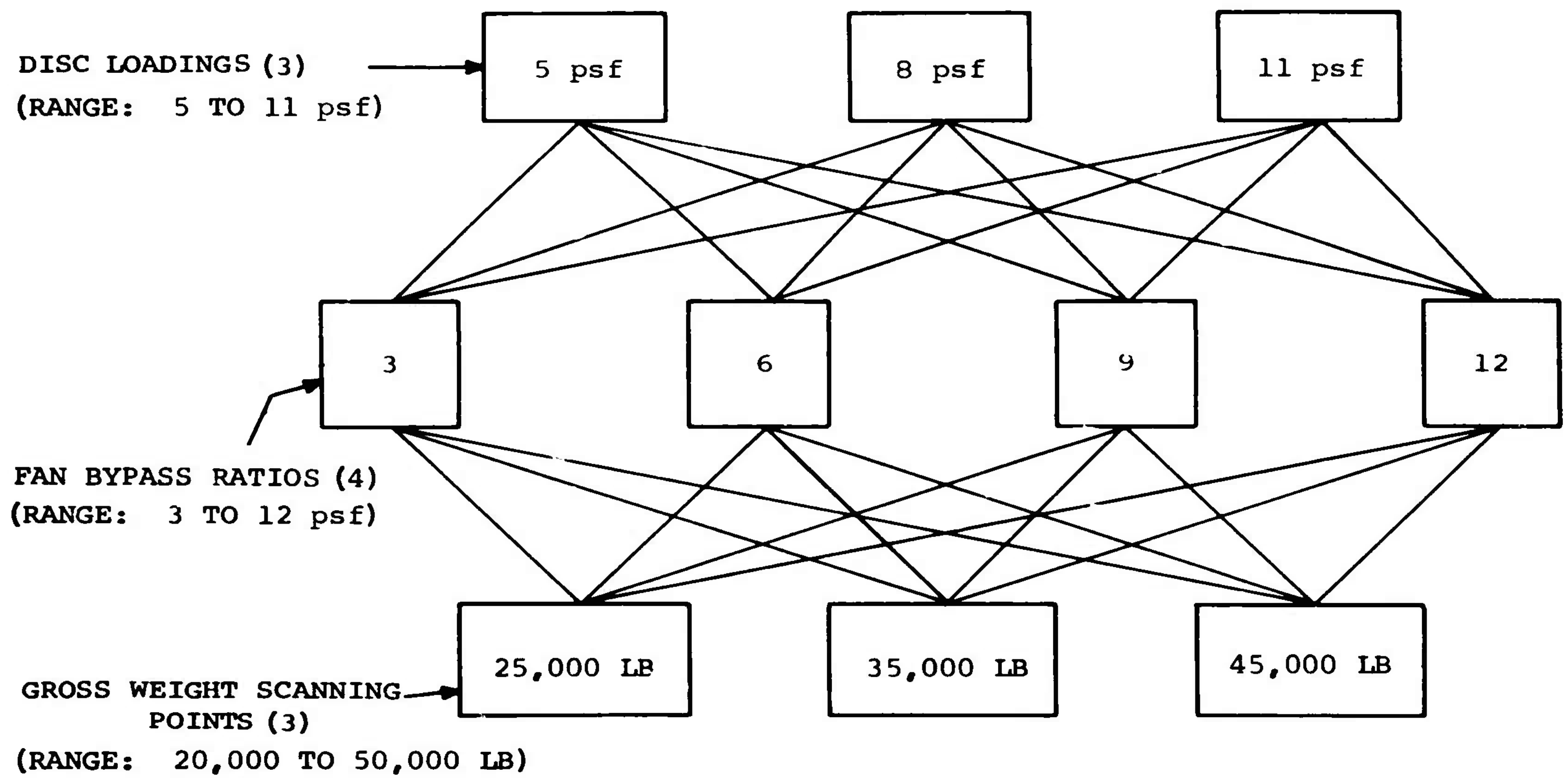


Figure 78. Matrix of 36 Major Parameters

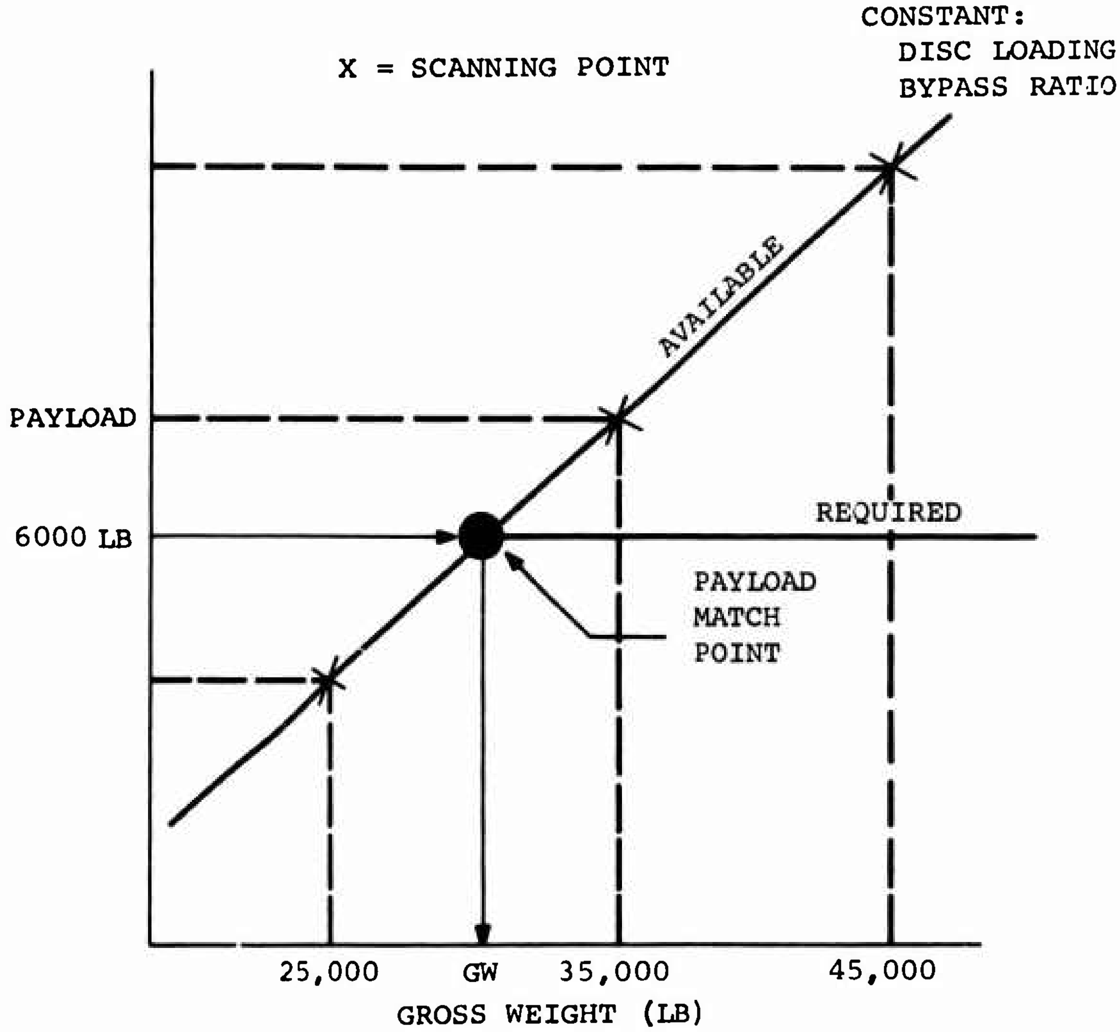


Figure 79. Gross Weight Determination for a Candidate Aircraft

The relative productivity values for the 12 candidate aircraft in each optimization case were then computed; the results were plotted against the matrix of disc loadings and bypass ratios, as shown in Figure 80. Aircraft gross weight variations were also shown for reference purposes.

In the optimization procedure the candidate aircraft with peak relative productivity within the parametric range of study was selected regardless of whether this aircraft had a minimum gross weight. If at a point in the procedure it became clear that certain parametric combinations would yield far-from-optimum candidate aircraft on the basis of productivity, these combinations were eliminated to reduce final computation loads. It was found possible in the selection of the optimum aircraft to pick integral values of disc loading and bypass ratio without compromising accuracy of the results. Discretion was employed in the selection with respect to practical aircraft design, and thus in some cases a slightly lower productivity aircraft with a moderate size rotor diameter was chosen instead of a higher productivity candidate with an extremely large rotor diameter.

Figures 81 through 110 show the results of the optimization procedure and indicate the candidate aircraft selected in each of the 30 cases studied. Tables V through X give a tabular summary of the results obtained directly from the selection plots. The design cruise speeds shown in this summary were obtained by interpolation of the aerodynamic data computed for the gross weight scanning points used in the analysis, and at the proper values of the other major parameters.

Selection of the optimum aircraft in terms of major parameters permitted design layout and group weight breakdown definition of these vehicles. Design data sheets were employed to define each aircraft for layout sizing purposes. The scaling data defined in the propulsion and airframe preliminary design phases of work was used for this purpose.

Aircraft summary weight statements (refer to Tables XII through XVII) and a further breakdown of propulsion system weights (refer to Tables XVIII through XXVII) for each optimized aircraft were also derived by use of the propulsion and airframe weights scaling data at the specific parameter values of these aircraft. This weight data is discussed in the following paragraphs.

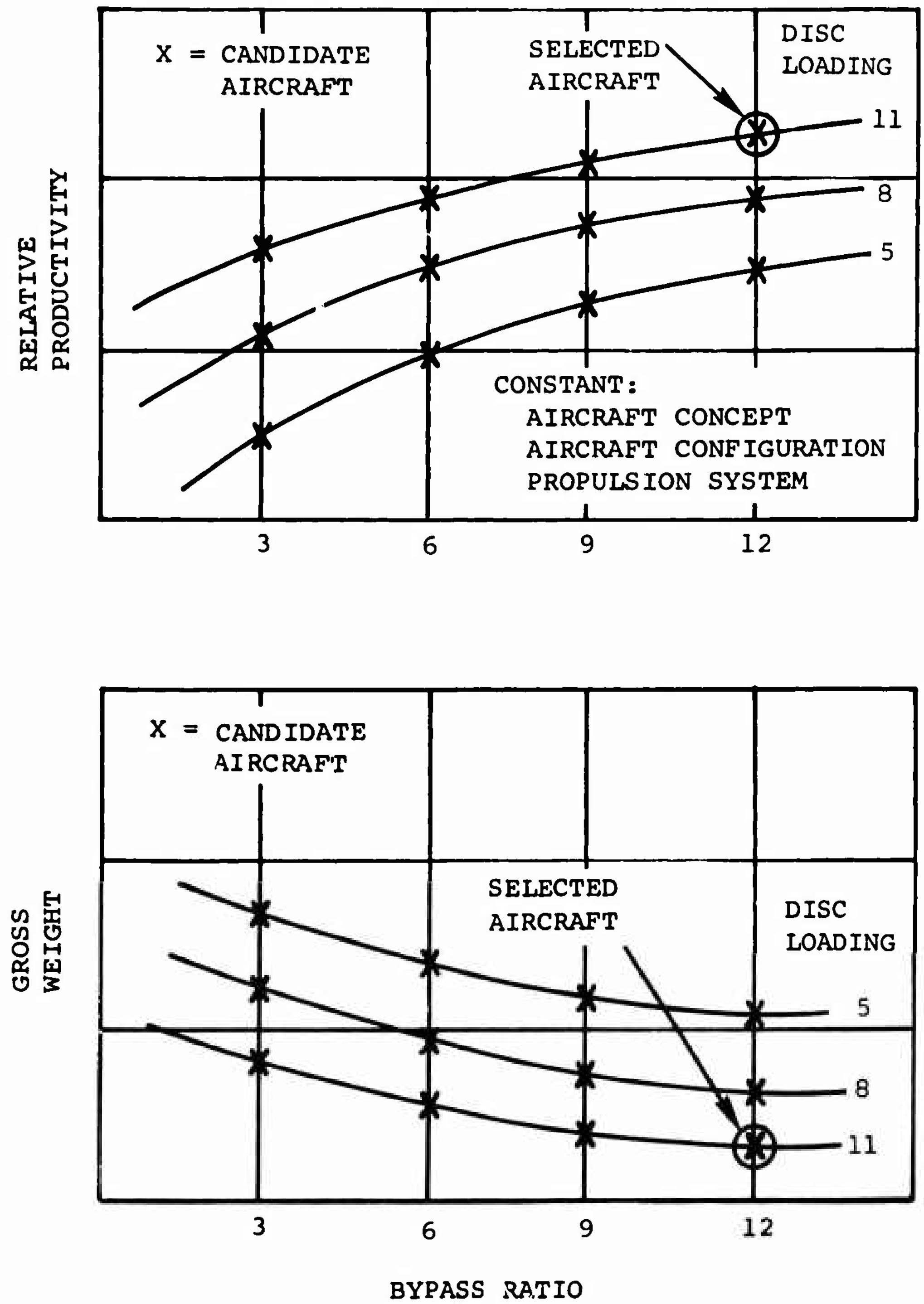


Figure 80. Determination of an Optimum Aircraft

Weight Estimating Procedures

The weights presented in this study were derived from previous experience of The Boeing Company in terms of statistical and analytical trends, from parametric data contained in the cruise fan propulsion system study of Reference 1, or from specific established ground rules.

Inputs for determination of weight scaling trends of airframe groups were obtained from the design layout drawings and from design scaling rules set forth elsewhere in this report. Additional inputs were obtained from an aerodynamic analysis performed over the matrix of parametric aircraft studied and from the propulsion analysis of the scaling factors involved in the five propulsion systems studied, in which propulsion item weights varied as functions of a design airflow parameter. Ground rules established the weights of fixed equipment and fixed useful load items for all parametric aircraft under consideration.

Group weight data sheets were prepared from these inputs based on the format of MIL-STD-451, Part 1, covering all aircraft type and propulsion system combinations in the required disc loadings and fan bypass ratios selected for study. For these combinations a gross weight scanning range of 20,000 to 50,000 pounds was established, and weight summaries were prepared for specific gross weight points of 25,000, 35,000 and 45,000 pounds in each case. This weight data provided the basis for the selection process of candidate-for-optimum and the final-optimum aircraft.

Selection of the optimum aircraft allowed use of the same parametric weight trend data to be employed in defining the aircraft group weight summaries and propulsion system weight breakdowns presented in this report.

Fundamental Ground Rules - Weight items held constant on a ground-rule basis for all aircraft were: fixed equipment at 2935 pounds, fixed useful load at 750 pounds, and design mission payload at 6000 pounds. Other design constants used in the estimation of weights are as follows:

Rotor tip speed	750 fps factored by 1.2 for design limit
Rotor blade t/c	15 percent at 0.25 radius

Rotor blade average t/c	11 percent
Limit load factor	3.0
Ultimate load factor	4.5
Wing aspect ratio	6
Wing t/c	15 percent
Taper ratio	0.5 or 0.35, depending on configuration

<u>1970 State of the Art</u> - The engine and cruise fan weights used in this study were based on the data contained in Reference 1. The weight data given in this report is projected for the 1970 time period.

The structural and drive system group weights were derived from the current Boeing Company weight trends. Adjustments were made to these weights for the improvements in material and manufacturing technologies anticipated by 1970.

The structural group items were reduced by 5 percent to allow for projected greater use of titanium, honeycomb sandwich construction, and advanced rotor hub and blade concepts.

The drive system weights were reduced by 10 percent to allow for increased hertz stress index resulting from gear improvements through shot-peened surfaces, increased carbon content, and for new steels and lubricants.

The percentage weight reductions anticipated are considered to be conservative and within the limits of known improvements in technology.

<u>Weights Procedure</u> - Using the input data supplied by the technology groups, weights were evaluated for each parametric aircraft point in accordance with the trends previously discussed. For each aircraft type and disc loading, weights were assessed at the three scanning design gross weight values and the four bypass ratios. In each case the weight empty, fixed useful load, and mission fuel were added and their total subtracted from the gross weight to obtain an available payload. These payload available figures were then plotted against a gross weight scale to determine the gross weight of the aircraft

capable of carrying the required 6000-pound payload as previously indicated.

Based on the method described in the preceding section, optimum aircraft were selected. For each case the gross weight, disc loading, and bypass ratio of these optimum aircraft were then defined. The group weight statements for these aircraft were made up by means of interpolation between appropriate points within the parametric range of variables.

During the early stages of aircraft design layout investigation, center-of-gravity location and balance checks were conducted on several configurations. These preliminary checks gave the design group information necessary to plan and to lay out the final designs for the optimum aircraft.

Supplemental studies involving high-speed sensitivity of aircraft using a typical integrated propulsion system and an aircraft employing tilting cruise fans were also performed. The weight information required in these cases was gained by applying the same weight trend data and by using the same weight ground rules employed in the basic study.

OPTIMUM AIRCRAFT DESIGN DEFINITION

A previous section of this report discussed the results of airframe-propulsion design integration studies conducted on a generalized basis prior to determination of the optimum aircraft for each case. These layout investigations were made with an assumed midrange design gross weight of 35,000 pounds in all cases. The studies yielded selections of basic locations and configurations of propulsion systems with respect to the various airframe layouts, and they provided the necessary design guides for scaling the aircraft as required for the parametric study.

This section presents the major physical design characteristics of the resulting optimum aircraft of the study. Three-view drawings of the optimum aircraft are shown in Figures 111 through 127. The principal parameter values of each aircraft are called out on these drawings. Propulsion system component locations and arrangements on the aircraft are those selected in the general investigations noted above, and the comments as to design selections and installation factors made previously pertain here for all optimum aircraft cases. Specific design points and problem areas arising with the optimum aircraft

selections shown on the three-view drawings are also discussed. These are taken up by propulsion system category. With one exception three-view drawings of propulsion-unloaded aircraft are not presented. Review of the optimization study results for propulsion-unloaded types showed that most of those having integrated powerplant systems required very small fans.

Fan sizes for the low design airflows involved were well below the accurate parametric scaling range of the powerplant data used in the study; therefore, no firm basis for a propulsion installation drawing existed. Comparison of the desired design configurations of powerplant systems for these aircraft types with those of the equivalent lift/propulsion-unloaded machines showed that no essential differences were involved. The only major difference from an overall aircraft viewpoint was the absence of a wing in one of the types. This factor allowed greater flexibility in vertical location of the cruise fans on tandem rotor aircraft without concern for possible effects of wing wake on fan inlets. All essential features of integrated powerplant systems for propulsion-unloaded aircraft are, therefore, shown in the drawings presented for the winged compound types. The shaft-coupled system 2a is presented in Figure 115 to illustrate the approximate size of the small fans on a typical tandem rotor propulsion-unloaded aircraft.

In the case of propulsion-unloaded aircraft with independent cruise fans, the optimum single and tandem rotor types resulting from the ground rules were impractical aircraft in terms of rotor geometry; therefore, three-view drawings of them are not presented. In these cases it was attempted to size the independent cruise fans for a very high maximum speed of 270 knots to assess the practicality of such an aircraft. The result was a requirement for an extreme amount of rotor blade area. In the tandem rotor case, for example, the resulting area requirement was about 1,100 square feet, which gave a blade chord of 6.5 feet with three-blade rotors 56.7 feet in diameter. This corresponds to a blade aspect ratio of 4 at the optimum disc loading of 8. Even if rotor overlap considerations allowed a greater number of blades per rotor, the chord requirements would be excessive. The large blade area requirement resulted from the combination of the high-speed goal established and imposition of a limiting rotor tip Mach number of 0.92.

At the high aircraft speed the rotor rpm must therefore be low; however, the weight of the aircraft must be supported, which

results in the necessity for extremely large blade areas. Very large cruise fans are also required, and the combination of rotor and fan requirements made both single and tandem rotor aircraft extremely heavy and impractical with a low relative productivity regardless of the high speed. As presented in the summary data of the optimization studies, the lift system characteristics of propulsion-unloaded aircraft with independent systems will not meet the design hover condition. Approximately 20 percent more installed shaft turbine power and additional lift system weight would be required to meet the 6000-foot, 95°F hover requirement when using the required large rotor blade area. Since the two aircraft concepts were impractical, a complete design analysis was not performed.

Gas Coupled Propulsion Systems 1a and 1b

Aircraft incorporating gas-coupled systems are shown in Figures 111, 112, 113, and 114. All drawings portray designs using the 1a systems with tip-driven cruise fans. The 1b systems with hub-driven fans are not shown because the differences in the two systems were insignificant for three-view layout purposes. Also, their basic vehicle parameters were quite similar.

The optimum tandem rotor lift/propulsion-unloaded aircraft of Figure 111 has no rotor overlap due to a combination of moderate gross weight and high disc loading. Minimum internal length requirements of the aircraft govern the rotor mast separation. Placement of the common power turbine in the thickest portion of the aft pylon and the desire for very short gas duct lengths governed the positioning of the gas-coupled propulsion components. The wing was not tilted because almost no portion of the wing area beneath the rotor discs lies outside the landing gear sponson plan area or the deflecting wing flap section. In general, the arrangement appears to be feasible.

The optimum tandem rotor composite aircraft shown in Figure 112 has the same rotor mast separation and the same governing internal length requirements. The rotor diameter selection, based on optimum gross weight and disc loading, results in a small amount of rotor disc area overlap, and the wing is shown as a tilting type. However, the choice of tilting was borderline in this case.

The gas-coupled propulsion system layout was based on the same considerations noted above. In this case, however, the gas

generator inlets are completely masked in cruise flight by the bulk of the forward rotor fairing, and a serious question arises as to whether the inlet system is practical. Raising the gas generators and moving them forward to reduce this masking could be accomplished at a penalty in gas duct length and with body fairing problems. Also, location of the retracted rear rotor fairing places a limit on raising propulsion items. A more detailed study of the entire propulsion arrangement would have to be conducted to determine whether a clearly feasible arrangement could be effected.

The single rotor lift/propulsion-unloaded aircraft is depicted in Figure 113. Gas ducting and the common power turbine are buried in the forward portion of the rotor pylon, with gas generators being housed in a forward extension of this body. This arrangement appears to be very feasible.

The single rotor composite aircraft shown in Figure 114 has the same general propulsion system arrangement. Location of the fans does not appear to be compromised by the retracted rotor fairing position.

Remote Shaft-Coupled Propulsion System 2a

Aircraft with this system appear in Figures 115, 116, 117, 118, and 119. The optimum tandem rotor propulsion-unloaded aircraft of Figure 115 displays the basic propulsion system arrangement employed and approximates the small fan size required. The drawing was made specifically to illustrate this fan size which is generally representative of the results for other integrated systems in this type of aircraft. The drawing also illustrates the solution employed to eliminate possible engine exhaust impingement or a fan ingestion problem. Fans are raised above the shaft turbines and turbine exhaust is directed partially downward. In general, the propulsion arrangement appears to be practical.

The optimum tandem rotor lift/propulsion-unloaded aircraft presented in Figure 116 has no rotor overlap, and as a result the wing is fixed to the fuselage, since it was estimated that tilting would not substantially reduce rotor download on the wing. The shaft turbines are housed in a forward extension of the rear rotor pylon, and the connecting gearboxes are buried within the pylon lines. Since the cruise fans are larger in diameter than those of the propulsion-unloaded aircraft, the relative vertical displacement of engines and fan centerlines

is greater to keep engine exhaust clear of the fans. The synchronizing shaft runs above the wing carry-over structure for easy accessibility. In general, the propulsion installation was rated as being practical.

The equivalent tandem rotor composite aircraft is shown in Figure 117. In this configuration the wing tilts to alleviate the download from rotor wash in vertical flight. This configuration is similar to the previous case except for the size of the components; however, in cruise flight the shaft turbine inlets are again completely masked by the forward rotor fairing regardless of an effort to raise the engines by means of a vee cross-shaft arrangement. This arrangement presents a special inlet problem due to possible flow detachment or boundary-layer buildup; additional detailed study would be required to determine its practicability.

The single rotor lift/propulsion-unloaded aircraft of Figure 118 incorporates shaft turbines and cross-shafting within a forward extension of the main rotor pylon with fans located at the ends of the vee cross-shaft. The vee arrangement is used to lift the fans above the turbine exhaust pipe level. The propulsion layout appears to be acceptable if exhaust impingement on fans and fan reingestion problems can be completely eliminated.

The single rotor composite aircraft depicted in Figure 119 shows again the problem of shaft turbine exhaust with respect to cruise fan location. In this case the fans could not be raised without interfering with a retracted main rotor fairing. Either the whole rotor system could be raised, or the shaft turbine exhausts could be extended to clear the fans. The latter approach was selected, but the exhaust layout shown is not considered to be a good one.

Convertible Cruise Fan System 2b

The vehicles employing convertible cruise fans are shown in Figures 120, 121, 122, and 123.

The tandem rotor lift/propulsion-unloaded and composite aircraft using this propulsion system are illustrated in Figures 120 and 121 respectively. These drawings show that the propulsion systems are simple in layout and almost identical except for the size of the components. Rotor and wing geometry are such that the wing is tilted for vertical operation in the

composite type to minimize hover download, but it remains fixed on the winged compound aircraft. Since the convertible powerplant system is in one package, there are no relative positioning problems of inlets and exhausts. A vee cross-shaft is employed to locate the fans at the desired vertical location. The combining gearbox of the rotor drive system is buried in the forward base of the rear rotor pylon. In the composite aircraft the fans can be located so that no inlet masking is produced by the forward rotor fairing. Considerable freedom in the longitudinal location of fans is permitted to achieve an optimum center of gravity position without substantially disturbing rear pylon lines. These configurations are considered to be feasible and are particularly attractive from an installation viewpoint.

The equivalent single rotor aircraft employing convertible cruise fans are shown in Figures 122 and 123 for lift/propulsion-unloaded and composite types, respectively. These similar concepts are simple and appear to be feasible. In addition, they are attractive concepts from propulsion system installation aspects.

Independent Cruise Fan System 3

Vehicles incorporating cruise fans completely independent of the lift system, and thus employing four gas generators, are shown in Figures 124, 125, 126, and 127. The optimum tandem lift/propulsion-unloaded aircraft is illustrated in Figure 124. Since the wing is almost completely beneath the rotor blade overlap area, it is a tilting design to minimize hover download. The twin-shaft turbine system employed for lifting is conventional. The rather large independent cruise fans, resulting from a selection of 270 knots maximum speed, are located just outboard of the shaft turbines. It is believed that shaft turbine accessibility problems would result, and the overall powerplant installation is rated as rather poor for this reason.

The equivalent tandem rotor composite aircraft is shown in Figure 125. The wing is again tilted to minimize download because of the relative rotor-wing geometry resulting from the parametric study. In this aircraft the shaft turbine inlet masking problem cited previously is still present, and the potential shaft turbine accessibility problem due to cruise fan proximity still applies. In this case considerable difficulty was incurred in attempting to design shaft turbine

exhaust nozzles which would avoid impingement directly on the cruise fans. Vertical location of cruise fans was limited by considerations of wing wake disturbance on inlets and interference with a retracted rear rotor fairing. The overall installation is considered to be a poor one.

The single rotor lift/propulsion-unloaded and composite aircraft with independent systems, depicted in Figures 126 and 127, respectively, are very similar propulsion systems. Problems in both cases were fan ingestion of shaft turbine exhaust and shaft turbine exhaust impingement on the fans. Again, a problem of shaft turbine accessibility could result from the side-by-side location of four engines.

TABLE VI
CHARACTERISTICS OF TANDEM ROTOR
PROPULSION-UNLOADED COMPOUND AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading	11	11	8	8	8
Fan bypass ratio	12	9	9	12	9
Rotor diameter (ft)	37.6	37.4	43.4	44.2	56.7
Wing area (ft^2)	0	0	0	0	0
Installed SHP, SL	5,150	5,110	4,640	4,750	8,300
Installed fan thrust, (lb)	2,965	2,645	2,210	7,850	18,300
Cruise speed, SL (kn)	206	206	199	198	262
Maximum speed, SL (kn)	210	211	203	203	270
Relative productivity*	82.2	83.6	81.3	78.5	54.2
Airframe weight (lb)	8,940	8,868	8,558	8,768	17,725
Propulsion weight (lb)	3,155	2,987	3,157	3,447	8,370
Fixed Equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	15,030	14,790	14,650	15,150	29,030
Fixed useful load (lb)	750	750	750	750	750
Payload	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,470	2,460	2,300	2,700	4,720
Useful load (lb)	9,220	9,210	9,050	9,450	11,470
Design gross weight (lb)	24,250	24,000	23,700	24,600	40,500
Useful load η	38.0	38.4	38.1	38.4	28.3
Propulsion & fuel wt. (lb)	5,625	5,447	5,457	6,147	13,090
Ferry TOGW (lb)	34,200	34,000	35,800	36,900	60,800
Ferry range (n mi)	1,580	1,600	1,850	1,820	1,770
Ferry fuel (lb)	15,900	15,930	18,355	18,927	26,397

$$* \quad \text{Relative productivity} = \frac{\text{payload (lb)} \times V_{cruise}\ \text{(kn)}}{\text{empty weight (lb)}}$$

** Design mission - 100-n mi radius at SL; 10% reserves

TABLE VII
CHARACTERISTICS OF SINGLE ROTOR
PROPULSION-UNLOADED COMPOUND AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading	8	8	8	8	8
Fan bypass ratio	12	12	12	12	6
Rotor diameter (ft)	63.0	62.6	62.8	63.6	79.0
Wing area (ft^2)	0	0	0	0	0
Installed SHP, SL std	5,320	5,300	5,170	5,250	7,900
Installed fan thrust (lb)	3,710	3,710	3,280	8,660	16,450
Cruise speed, SL (kn)	212	212	212	212	266
Maximum speed, SL (kn)	216	216	213	216	270
Relative productivity*	80.7	82.1	80.9	78.8	58.5
Airframe weight (lb)	9,303	9,176	9,178	9,359	16,564
Propulsion weight (lb)	3,487	3,389	3,607	3,856	7,771
Fixed equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	15,725	15,500	15,720	16,150	27,270
Fixed useful weight (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,525	2,500	2,380	2,650	5,180
Useful load (lb)	9,275	9,250	9,130	9,400	11,930
Design gross weight (lb)	25,000	24,750	24,850	25,550	39,200
Useful load η	37.0	37.3	36.7	36.8	30.4
Propulsion & fuel wt. (lb)	6,012	5,889	5,987	6,506	12,951
Ferry TOGW (lb)	37,500	37,100	37,300	38,300	58,800
Ferry range (n mi)	1,990	1,980	1,950	1,925	1,840
Ferry fuel (lb)	19,450	19,200	19,710	19,710	28,475

*Relative productivity $= \dfrac{\text{payload (lb)} \times V_{cruise} \text{ (kn)}}{\text{empty weight (lb)}}$

** Design mission - 100-n mi radius at SL; 10% reserves

TABLE VIII
CHARACTERISTICS OF TANDEM ROTOR LIFT/
PROPULSION-UNLOADED COMPOUND AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading	11	11	11	11	8
Fan bypass ratio	12	12	12	9	3
Rotor diameter (ft)	39.4	39.8	40.6	39.9	49.6
Wing area (ft^2)	253	248	333	261	200
Installed SHP, SL std	5,730	5,890	6,260	6,000	6,400
Installed fan thrust (lb)	8,440	8,820	8,840	9,180	10,040
Cruise speed SL (kn)	219	223	201	219	259
Maximum speed, SL (kn)	225	230	209	226	270
Relative productivity*	75.9	74.8	64.5	73.6	76.0
Percent unload at V_{cruise}	83.0	83.2	86.0	83.5	78.0
Airframe weight (lb)	10,704	10,896	11,535	10,886	11,868
Propulsion weight (lb)	3,659	4,079	4,230	4,029	5,592
Fixed equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	17,298	17,910	18,700	17,850	20,395
Fixed useful load (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,642	2,670	3,050	2,900	4,135
Useful load (lb)	9,392	9,420	9,800	9,650	10,885
Design gross weight (lb)	26,690	27,330	28,500	27,500	31,280
Useful load η	35.2	34.5	34.4	35.1	34.8
Propulsion & fuel wt (lb)	6,301	6,749	7,280	6,929	9,727
Ferry TOGW (lb)	40,035	40,955	42,750	41,250	46,920
Ferry range (n mi)	2,000	1,930	2,080	1,960	1,950
Ferry fuel (lb)	20,157	20,551	21.460	20,860	23,805

* Relative productivity = $\dfrac{\text{payload (lb)} \times V_{cruise}\ \text{(kn)}}{\text{empty weight (lb)}}$

** Design mission - 100-n mi radius at SL; 10% reserves

TABLE IX

CHARACTERISTICS OF SINGLE ROTOR LIFT/PROPULSION-UNLOADED COMPOUND AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading (psf)	11	11	11	11	8
Fan bypass ratio	12	12	12	9	3
Rotor diameter (ft)	56.1	56.6	57.9	57.1	69.5
Wing area (ft^2)	225	221	289	225	194
Installed SHP, SL std	6,700	6,800	7,050	6,860	6,180
Installed fan thrust (lb)	9,860	10,200	9,960	10,530	9,880
Cruise speed, SL (kn)	234	237	216	237	260
Maximum speed, SL (kn)	240	244	223	243	270
Relative productivity*	79.1	77.6	63.1	77.0	78.9
Percent unload at V_{cruise}	82.0	81.7	84.8	81.6	78.2
Airframe weight (lb)	10,563	10,630	11,299	10,798	11,178
Propulsion weight (lb)	4,257	4,773	4,796	4,677	5,619
Fixed equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	17,755	18,338	19,030	18,410	19,732
Fixed useful load (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,895	2,712	3,250	3,260	4,038
Useful load (lb)	9,645	9,462	10,000	10,010	10,788
Design gross weight (lb)	27,400	27,800	29,030	28,420	30,520
Useful load η	35.2	33.0	34.4	35.2	35.4
Propulsion & fuel wt (lb)	7,152	7,485	8,046	7,937	9,657
Ferry TOGW (lb)	41,100	41,700	43,545	42,630	45,780
Ferry range (n mi)	2,110	2,030	2,130	2,080	1,940
Ferry fuel (lb)	20,805	20,902	21,905	21,640	23,378

$$*\text{Relative productivity} = \frac{\text{payload (lb)} \times V_{cruise} \text{ (kn)}}{\text{empty weight (lb)}}$$

**Design mission - 100-n mi radius at SL; 10% reserves

TABLE X

CHARACTERISTICS OF TANDEM ROTOR COMPOSITE AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading (psf)	11	11	11	11	8
Fan bypass ratio	12	9	12	6	3
Rotor diameter (ft)	43.9	44.2	45.0	44.9	54.9
Wing area (ft^2)	418	422	439	434	476
Installed SHP, SL std	7,900	8,000	8,270	8,180	7,920
Installed fan thrust (lb)	11,600	10,420	11,760	11,200	12,280
Cruise speed, SL (kn)	284	279	285	304	341
Maximum speed, SL (kn)	292	289	292	312	350
Relative productivity*	72.1	69.4	67.6	74.0	74.8
Airframe weight (lb)	15,806	15,892	16,563	16,271	17,448
Propulsion weight (lb)	4,914	5,323	5,812	5,417	6,992
Fixed equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	23,655	24,150	25,310	24,623	27,375
Fixed useful load (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,995	2,900	3,060	3,337	3,995
Useful load (lb)	9,745	9,650	9,810	10,087	10,745
Design gross wt (lb)	33,400	33,800	35,120	34,710	38,120
Useful load η	29.2	28.6	27.9	29.1	28.2
Propulsion & fuel wt (lb)	7,909	8,223	8,872	8,754	10,987
Ferry TOGW (lb)	50,100	50,800	52,600	52,000	57,100
Ferry range (n mi)	2,327	2,204	2,363	2,283	2,020
Ferry fuel (lb)	23,552	24,265	24,540	24,607	26,795

$$\text{*Relative productivity} = \frac{\text{payload (lb)} \times V_{cruise}\ \text{(kn)}}{\text{empty weight (lb)}}$$

**Design mission - 100-n mi radius at SL; 10% reserves

TABLE XI

CHARACTERISTICS OF SINGLE ROTOR COMPOSITE AIRCRAFT

Propulsion System:	1a	1b	2a	2b	3
Disc loading (psf)	11	11	11	11	8
Fan bypass ratio	12	9	12	6	3
Rotor diameter (ft)	61.0	62.3	62.3	62.3	75.5
Wing area (ft^2)	405	421	421	420	449
Installed SHP, SL std	8,030	8,300	8,210	8,180	7,370
Installed fan thrust (lb)	11,760	10,820	11,600	11,200	12,200
Cruise speed, SL (kn)	287	290	283	305	341
Maximum speed, SL (kn)	296	299	293	316	350
Relative productivity*	76.0	72.7	71.5	77.6	81.4
Airframe weight (lb)	14,761	15,173	15,165	15,048	15,425
Propulsion weight (lb)	5,054	5,754	5,650	5,597	6,808
Fixed equip weight (lb)	2,935	2,935	2,935	2,935	2,935
Empty weight (lb)	22,660	23,862	23,750	23,580	25,168
Fixed useful load (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)**	2,940	3,088	3,200	3,320	3,932
Useful load (lb)	9,690	9,838	9,950	10,070	10,682
Design gross wt (lb)	32,350	33,700	33,700	33,650	35,850
Useful load η	30.0	29.2	29.5	29.9	29.8
Propulsion & fuel wt (lb)	7,994	8,842	8,850	8,917	10,740
Ferry TOGW (lb)	48,500	50,500	50,500	49,600	53,800
Ferry range (n mi)	2,348	2,227	2,383	2,305	2,041
Ferry fuel (lb)	23,210	23,908	24,020	23,665	25,832

*Relative productivity = $\dfrac{\text{payload (lb)} \times V_{cruise}\text{ (kn)}}{\text{empty weight (lb)}}$

**Design mission - 100-n mi radius at SL; 10% reserves

TABLE XII. SUMMARY WEIGHT STATEMENT FOR TANDEM ROTOR PROPULSION-UNLOADED AIRCRAFT

GROSS WEIGHT	24,250	24,000	23,700	24,600	40,500
DISC LOADING	11	11	8	8	8
BYPASS RATIO	12	9	9	12	9
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	2,313	2,286	2,109	2,237	7,363
WING GROUP	-	-	-	-	-
TAIL GROUP	-	-	-	-	-
BODY GROUP	4,247	4,231	4,209	4,256	5,001
ALIGHTING GEAR	921	912	893	935	1,539
FLIGHT CONTROLS	1,330	1,315	1,225	1,283	3,610
ENGINE SECTION	129	124	122	57	212
PROPULSION GROUP					
ENGINES (S)	595	593	540	1,100	1,025
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	174	173	170	185	338
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	360	230	300	INCL WITH ENG	2,780
DRIVE SYSTEM	1,893	1,858	2,014	2,062	3,994
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	15,030	14,790	14,650	15,150	29,030
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,470	2,460	2,300	2,700	4,720
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	24,250	24,000	23,700	24,600	40,500

TABLE XIII. SUMMARY WEIGHT STATEMENT FOR SINGLE ROTOR PROPULSION-UNLOADED AIRCRAFT

GROSS WEIGHT	25,000	24,750	24,850	25,550	39,200
DISC LOADING	8	8	8	8	8
BYPASS RATIO	12	12	12	12	6
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	2,428	2,375	2,340	2,473	6,755
WING GROUP	-	-	-	-	-
TAIL GROUP	273	270	270	280	446
BODY GROUP	4,512	4,465	4,504	4,545	5,363
ALIGHTING GEAR	950	940	945	969	1,491
FLIGHT CONTROLS	1,010	1,000	988	1,030	2,309
ENGINE SECTION	130	126	131	62	200
PROPULSION GROUP					
ENGINES (S)	608	603	635	1,260	1,000
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	177	175	170	182	359
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	340	280	400	INCL WITH ENG	2,260
DRIVE SYSTEM	2,229	2,198	2,269	2,314	3,919
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	15,725	15,500	15,720	16,150	27,270
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,525	2,500	2,380	2,650	5,180
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	25,000	24,750	24,850	25,550	39,200

TABLE XIV. SUMMARY WEIGHT STATEMENT FOR TANDEM ROTOR LIFT/PROPULSION-UNLOADED AIRCRAFT

GROSS WEIGHT	26,690	27,330	28,500	27,500	31,280
DISC LOADING	11	11	11	11	8
BYPASS RATIO	12	12	12	9	3
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	2,222	2,290	2,410	2,309	2,726
WING GROUP	1,060	1,067	1,345	1,107	1,033
TAIL GROUP	242	237	318	249	200
BODY GROUP	4,370	4,408	4,455	4,403	4,798
ALIGHTING GEAR	1,014	1,038	1,082	1,045	1,189
FLIGHT CONTROLS	1,653	1,700	1,775	1,710	1,966
ENGINE SECTION	143	156	150	63	166
PROPULSION GROUP					
ENGINES (S)	659	679	770	1,400	840
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	182	183	202	194	311
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	570	920	690	INCL WITH ENG	1,230
DRIVE SYSTEM	2,115	2,164	2,435	2,335	2,983
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	17,298	17,910	18,700	17,850	20,395
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,642	2,670	3,050	2,900	4,135
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	26,690	27,330	28,500	27,500	31,280

TABLE XV. SUMMARY WEIGHT STATEMENT FOR SINGLE ROTOR LIFT/PROPULSION-UNLOADED AIRCRAFT

GROSS WEIGHT	27,400	27,800	29,030	28,420	30,520
DISC LOADING	11	11	11	11	8
BYPASS RATIO	12	12	12	9	3
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	2,347	2,365	2,527	2,460	2,546
WING GROUP	1,000	998	1,214	1,021	960
TAIL GROUP	375	370	453	402	353
BODY GROUP	4,370	4,388	4,494	4,451	4,636
ALIGHTING GEAR	1,041	1,056	1,102	1,081	1,158
FLIGHT CONTROLS	1,273	1,283	1,344	1,316	1,359
ENGINE SECTION	157	170	165	67	166
PROPULSION GROUP					
ENGINES (S)	760	770	860	1,625	865
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	194	186	210	211	305
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	650	1,155	770	INCL WITH ENG	1,210
DRIVE SYSTEM	2,520	2,529	2,823	2,741	3,006
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	17,755	18,338	19,030	18,410	19,732
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,895	2,712	3,250	3,260	4,038
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	27,400	27,800	29,030	28,420	30,520

TABLE XVI. SUMMARY WEIGHT STATEMENT FOR TANDEM ROTOR COMPOSITE AIRCRAFT

GROSS WEIGHT	33,400	33,800	35,120	34,710	38,120
DISC LOADING	11	11	11	11	8
BYPASS RATIO	12	9	12	6	3
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	4,275	4,295	4,560	4,503	4,940
WING GROUP	1,748	1,757	1,880	1,843	2,062
TAIL GROUP	397	401	416	413	446
BODY GROUP	5,767	5,785	5,900	5,862	5,919
ALIGHTING GEAR	1,269	1,283	1,334	1,318	1,448
FLIGHT CONTROLS	2,175	2,185	2,290	2,261	2,441
ENGINE SECTION	175	186	183	71	192
PROPULSION GROUP					
ENGINES (S)	898	900	1,050	1,810	1,070
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	198	194	201	215	304
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	855	1,250	1,020	INCL WITH ENG	1,640
DRIVE SYSTEM	2,830	2,846	3,408	3,292	3,745
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	23,655	24,150	25,310	24,623	27,375
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,995	2,900	3,060	3,337	3,995
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	33,400	33,800	35,120	34,710	38,120

TABLE XVII. SUMMARY WEIGHT STATEMENT FOR SINGLE ROTOR COMPOSITE AIRCRAFT

GROSS WEIGHT	32,350	33,700	33,700	33,650	35,850
DISC LOADING	11	11	11	11	8
BYPASS RATIO	12	9	12	6	3
PROPULSION SYSTEM	1a	1b	2a	2b	3
ROTOR GROUP	4,066	4,247	4,247	4,247	4,251
WING GROUP	1,677	1,767	1,767	1,767	1,919
TAIL GROUP	572	591	591	591	522
BODY GROUP	5,448	5,534	5,534	5,534	5,605
ALIGHTING GEAR	1,230	1,281	1,281	1,277	1,360
FLIGHT CONTROLS	1,501	1,563	1,563	1,563	1,582
ENGINE SECTION	177	190	182	71	186
PROPULSION GROUP					
ENGINES (S)	943	982	980	1,830	960
EXHAUST SYSTEM	33	33	33	-	33
FUEL SYSTEM	195	203	208	213	301
ENGINE CONTROLS	40	40	40	40	80
STARTING SYSTEM	60	60	60	60	120
FAN INST	840	1,350	1,000	INCL WITH ENG	1,640
DRIVE SYSTEM	2,943	3,086	3,329	3,454	3,674
AUX POWER PLANT	100	100	100	100	100
INSTR AND NAV	170	170	170	170	170
HYDR AND PNEU	225	225	225	225	225
ELECTRICAL GROUP	550	550	550	550	550
ELECTRONICS GROUP	285	285	285	285	285
ARMOR	300	300	300	300	300
FURN & EQUIP GROUP	860	860	860	860	860
AIR COND & DE-ICING	180	180	180	180	180
AUXILIARY GEAR	265	265	265	265	265
WEIGHT EMPTY	22,660	23,862	23,750	23,580	25,168
FIXED USEFUL LOAD	750	750	750	750	750
CREW	(600)				
TRAPPED LIQUIDS	(70)				
ENGINE OIL	(80)				
FUEL	2,940	3,088	3,200	3,320	3,932
PAYLOAD	6,000	6,000	6,000	6,000	6,000
GROSS WEIGHT	32,350	33,700	33,700	33,650	35,850

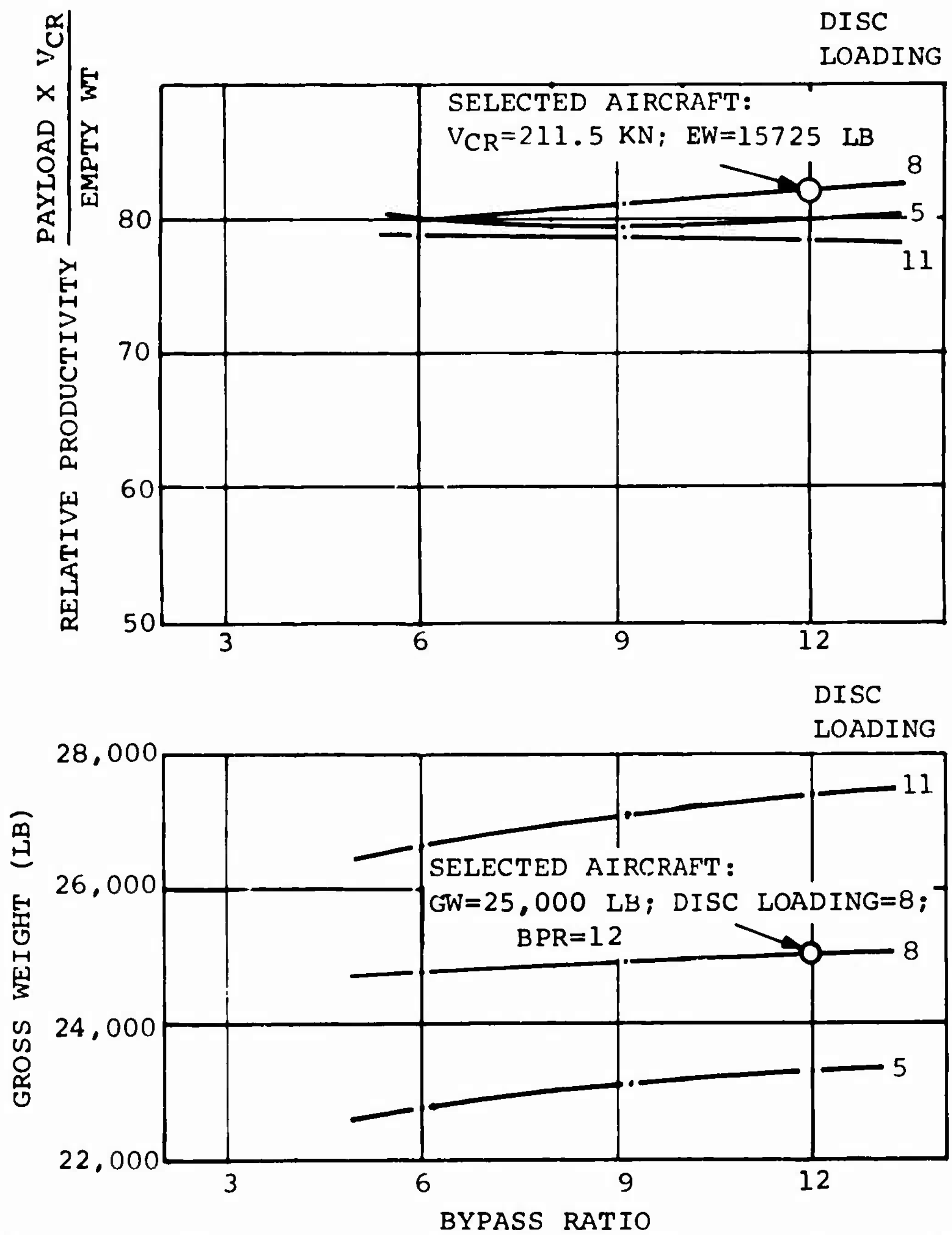


Figure 81. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Propulsion-Unloaded Aircraft Propulsion System 1a

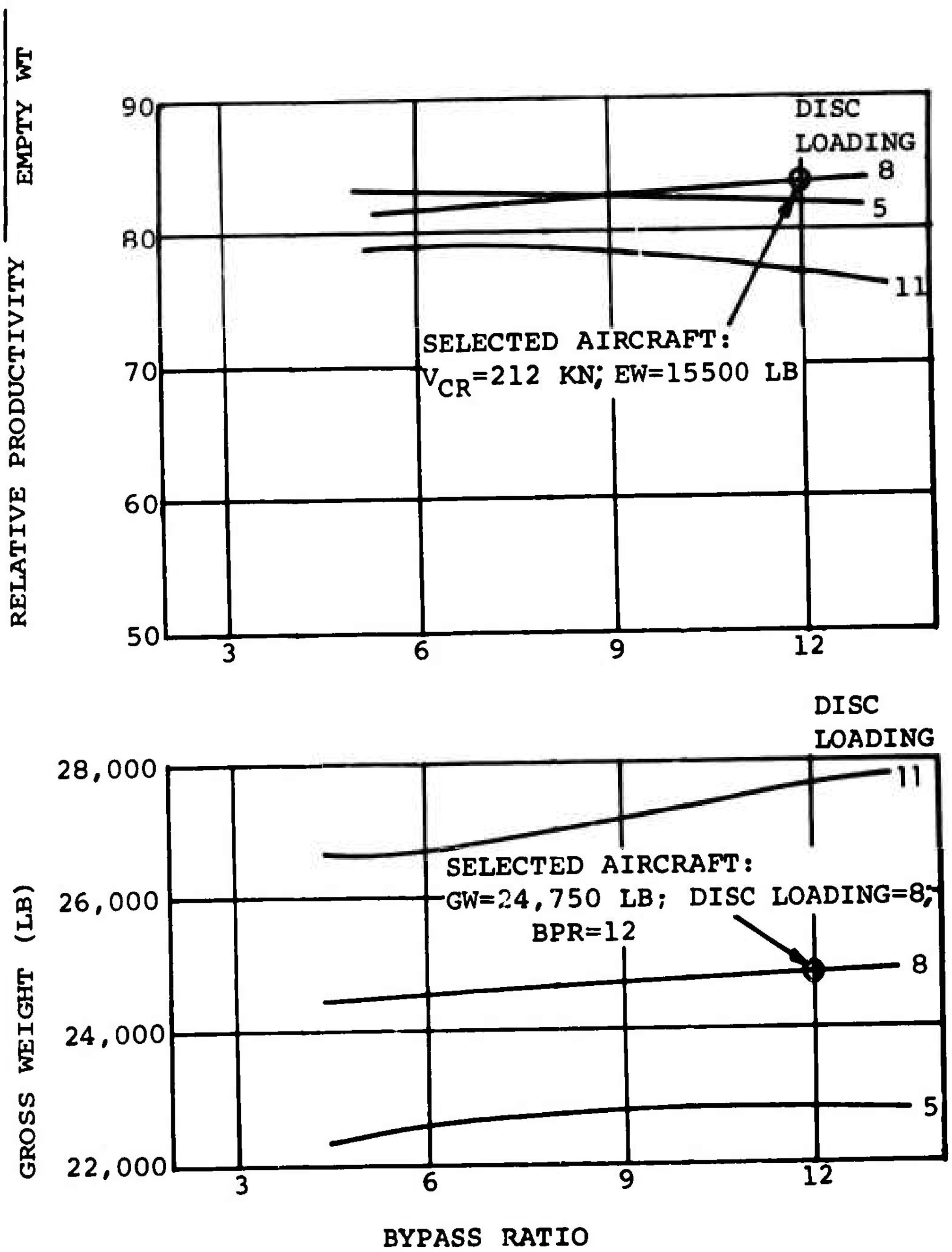


Figure 82. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio – Single Rotor Propulsion-Unloaded Aircraft Propulsion System lb

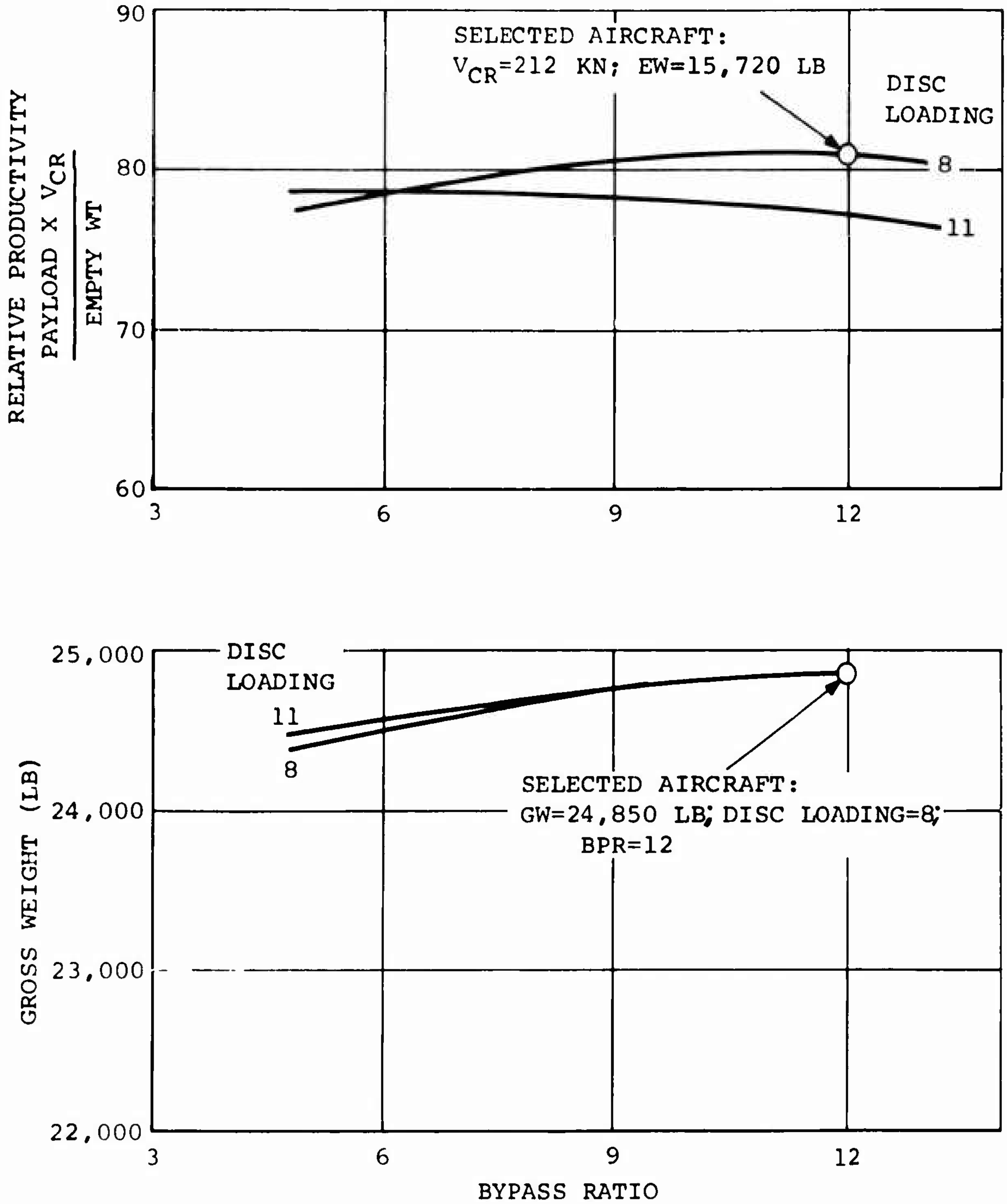


Figure 83. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Propulsion-Unloaded Aircraft Propulsion System 2a

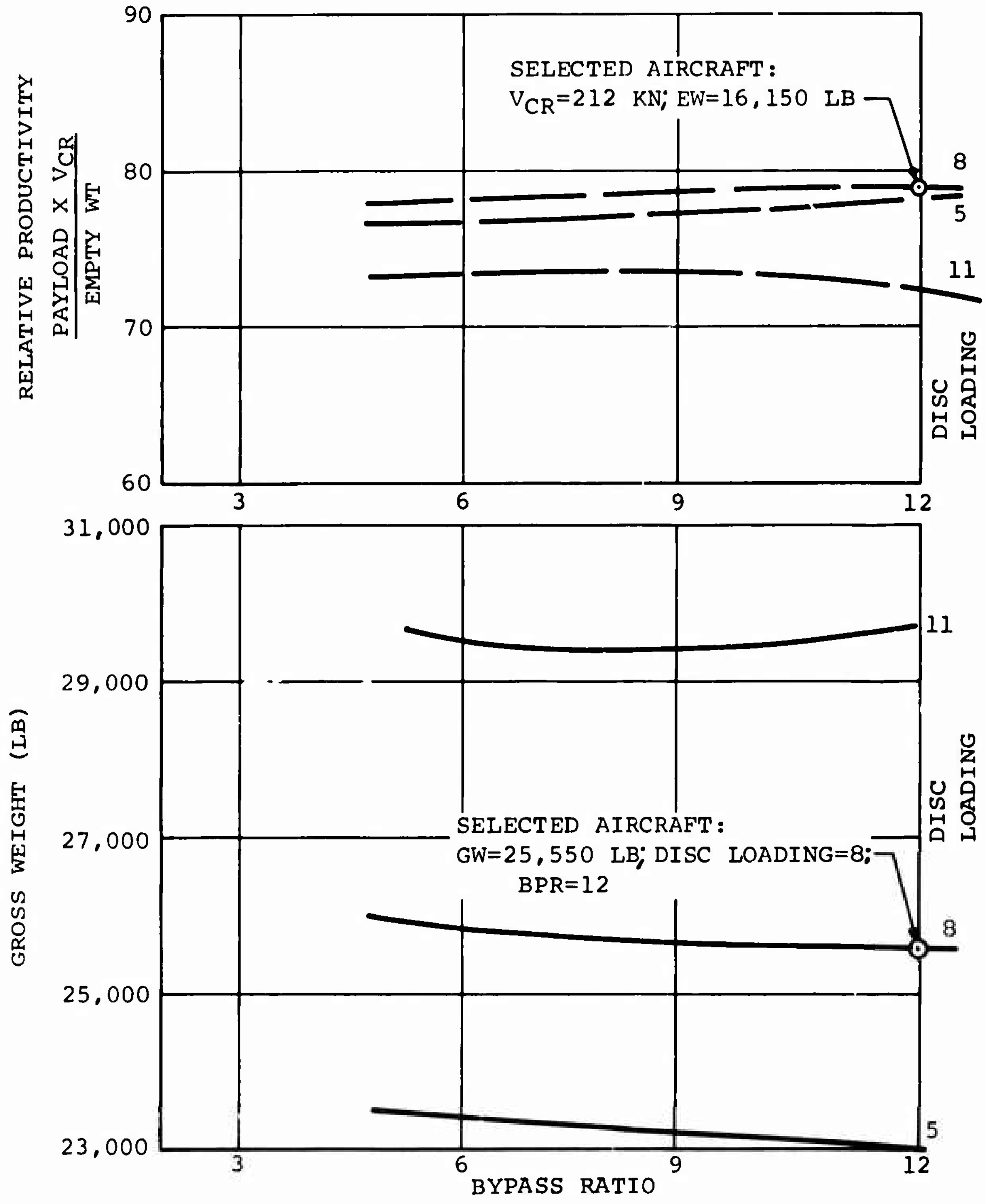
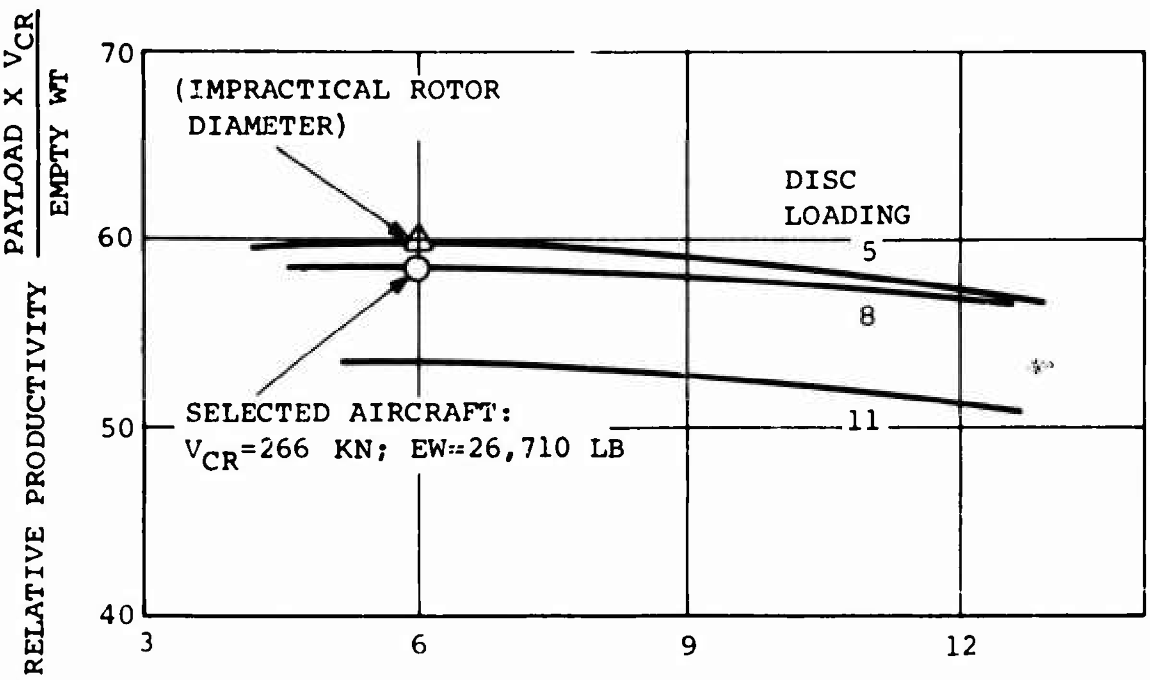


Figure 84. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Propulsion-Unloaded Aircraft Propulsion System 2b



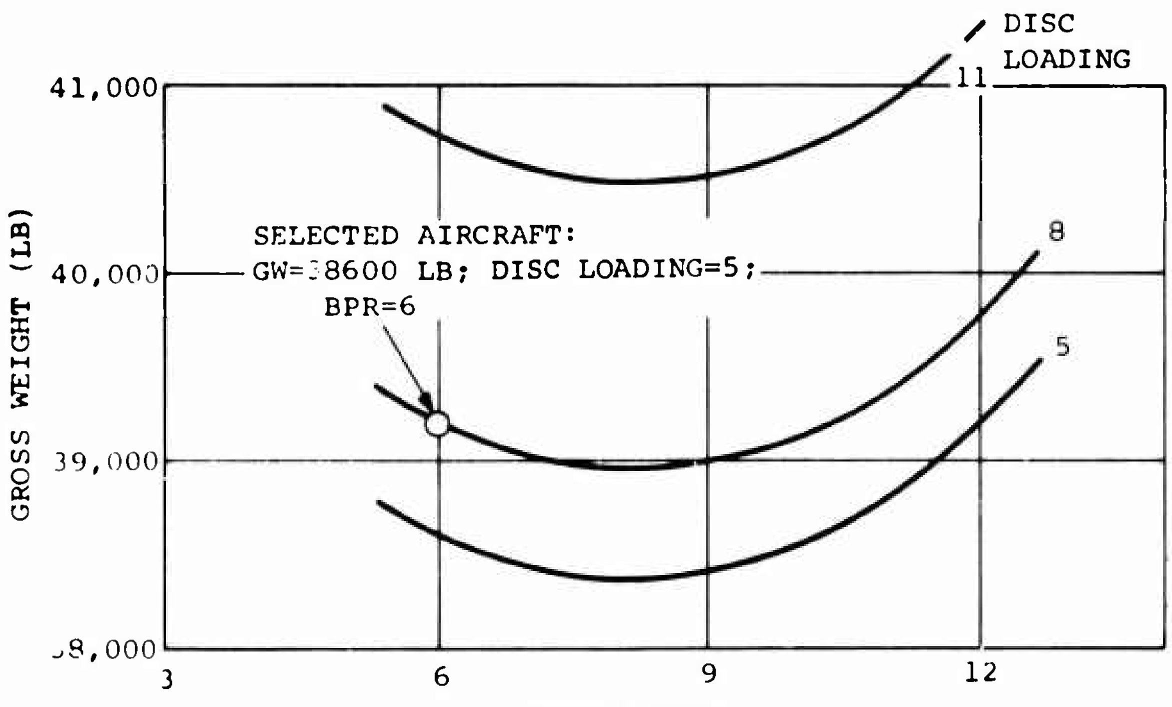


Figure 85. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Propulsion-Unloaded Aircraft Propulsion System 3

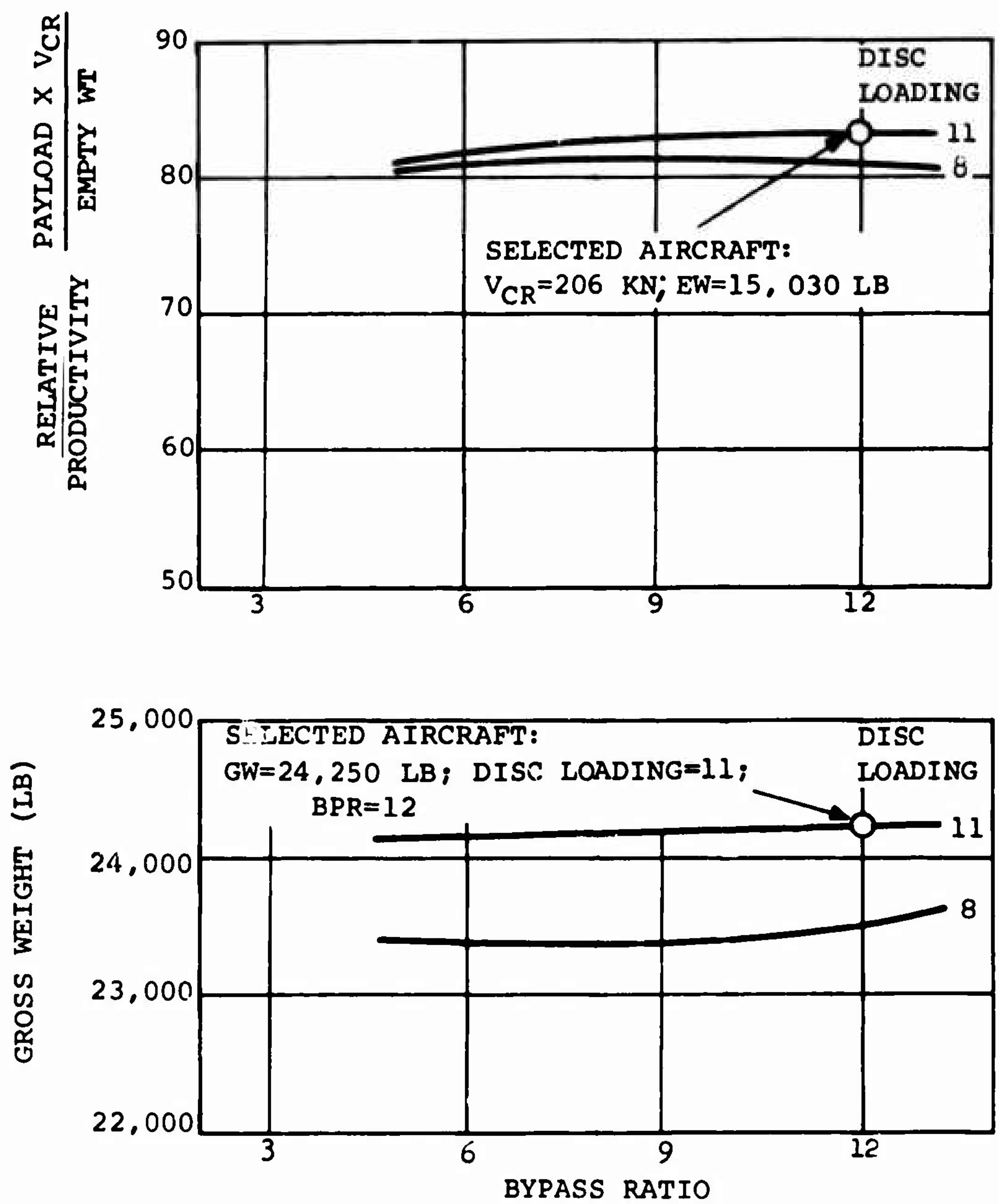


Figure 86. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Propulsion-Unloaded Aircraft Propulsion System 1a

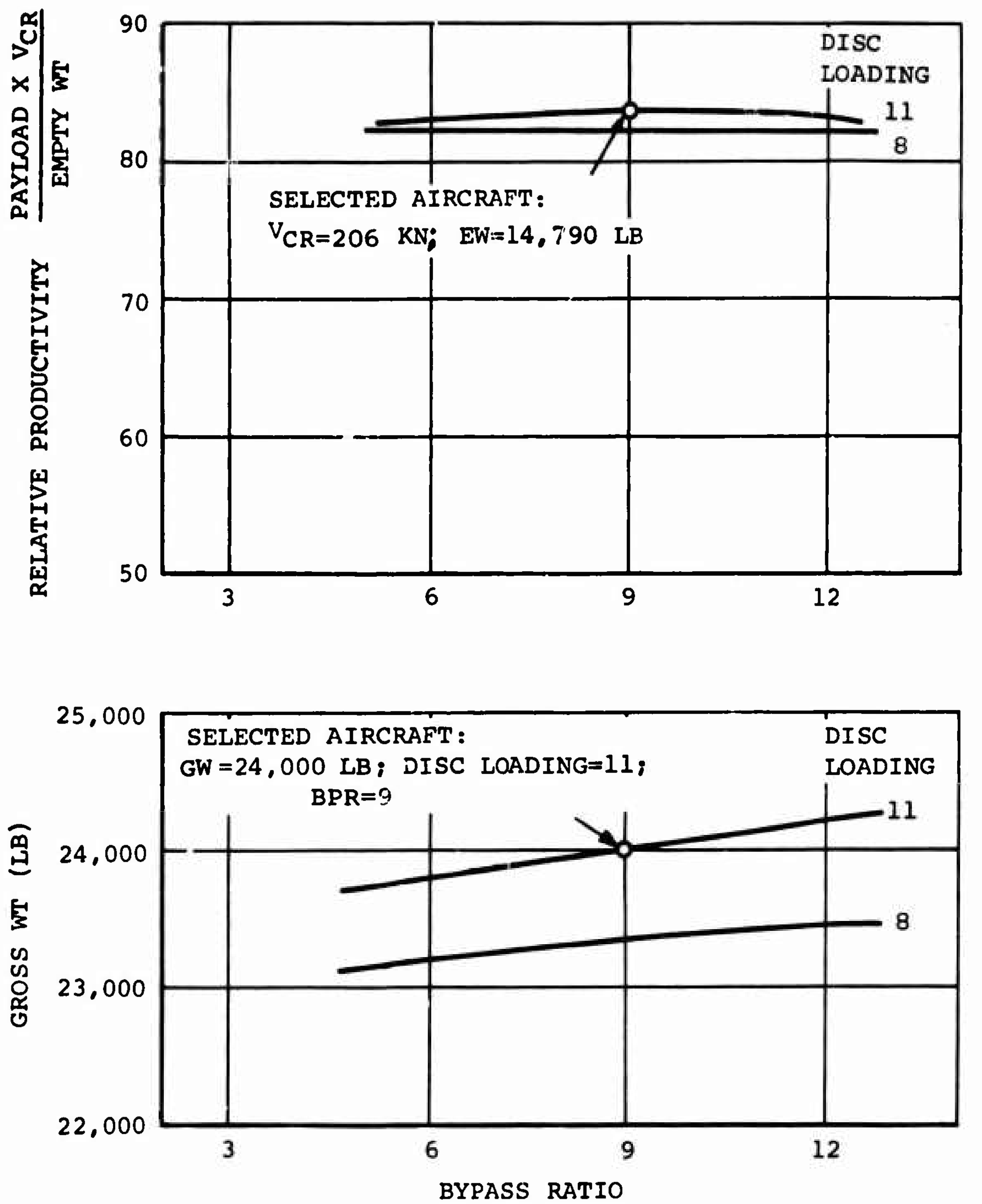


Figure 87. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Propulsion-Unloaded Aircraft Propulsion System 1b

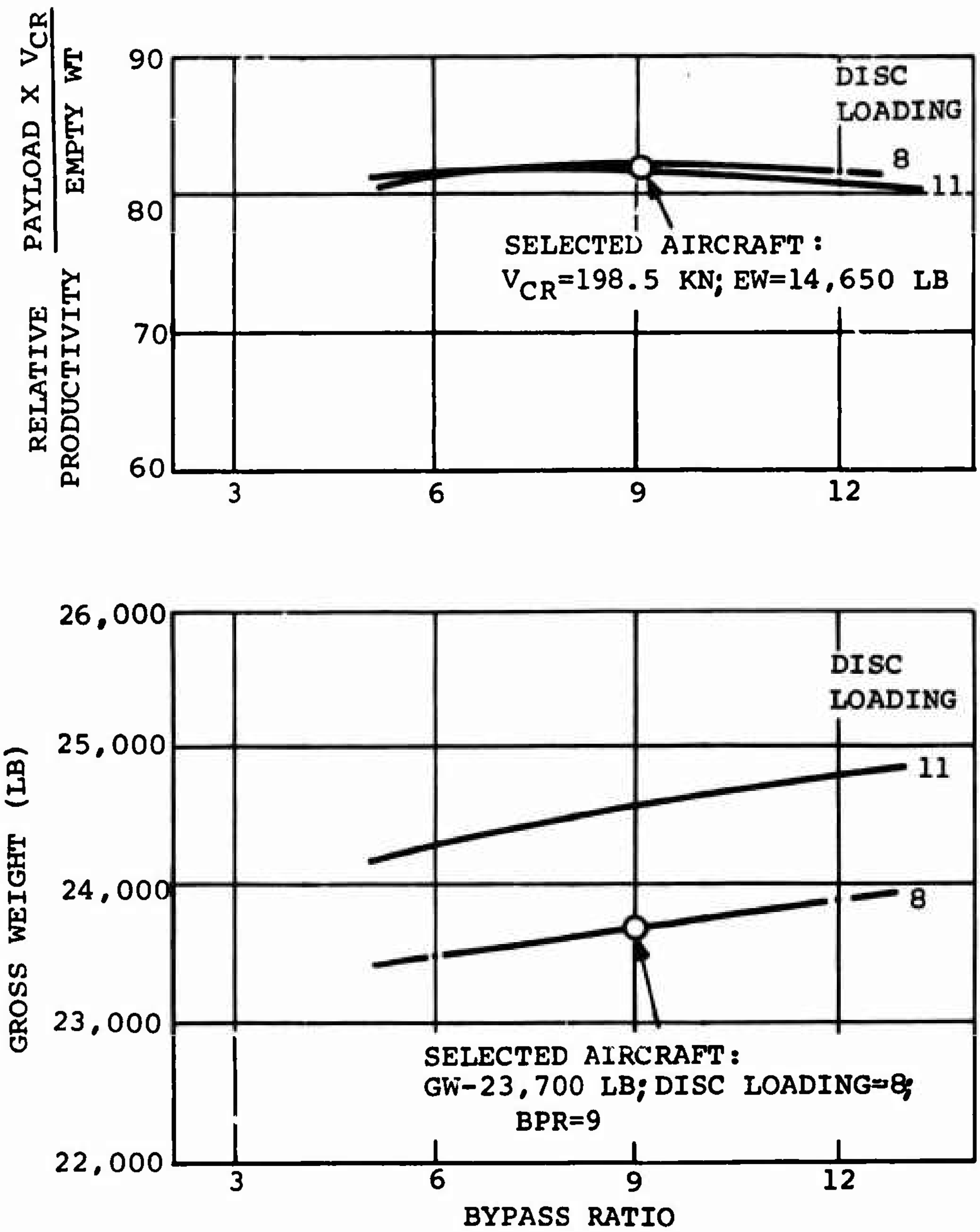


Figure 88. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio-Tandem Rotor Propulsion-Unloaded Aircraft Propulsion System 2a

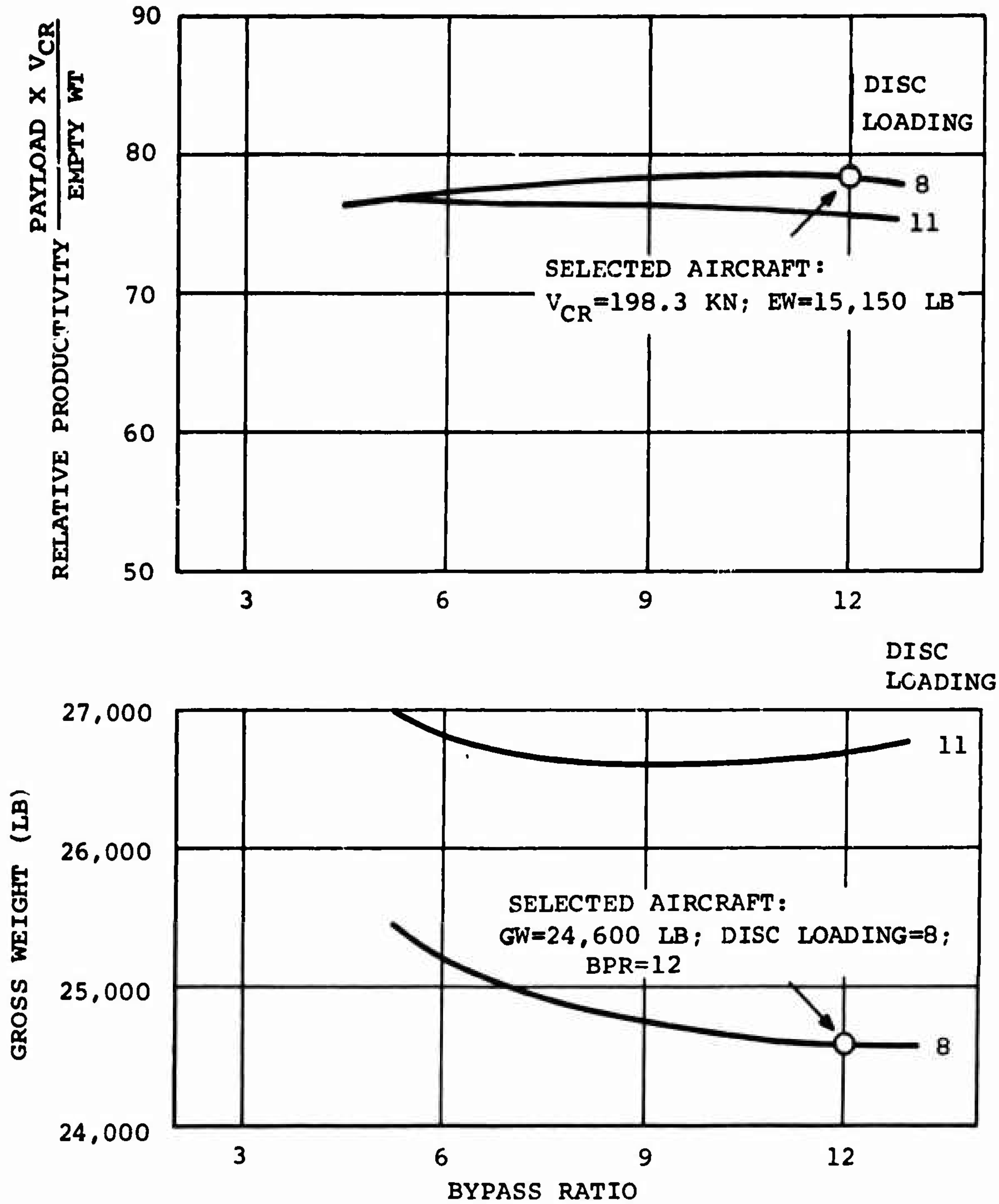


Figure 89. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Propulsion-Unloaded Aircraft Propulsion System 2b

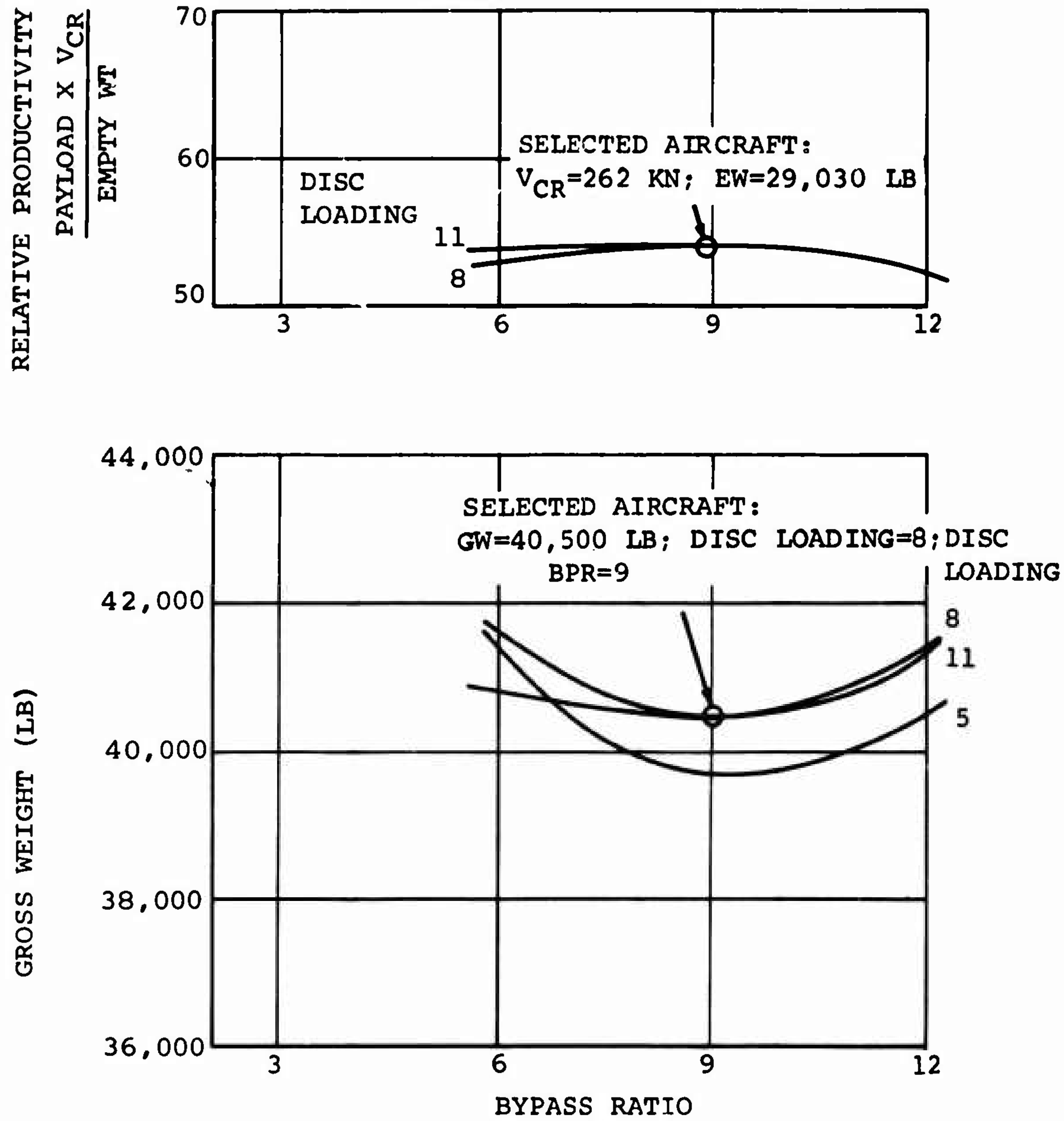


Figure 90. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Propulsion Unloaded Aircraft Propulsion System 3

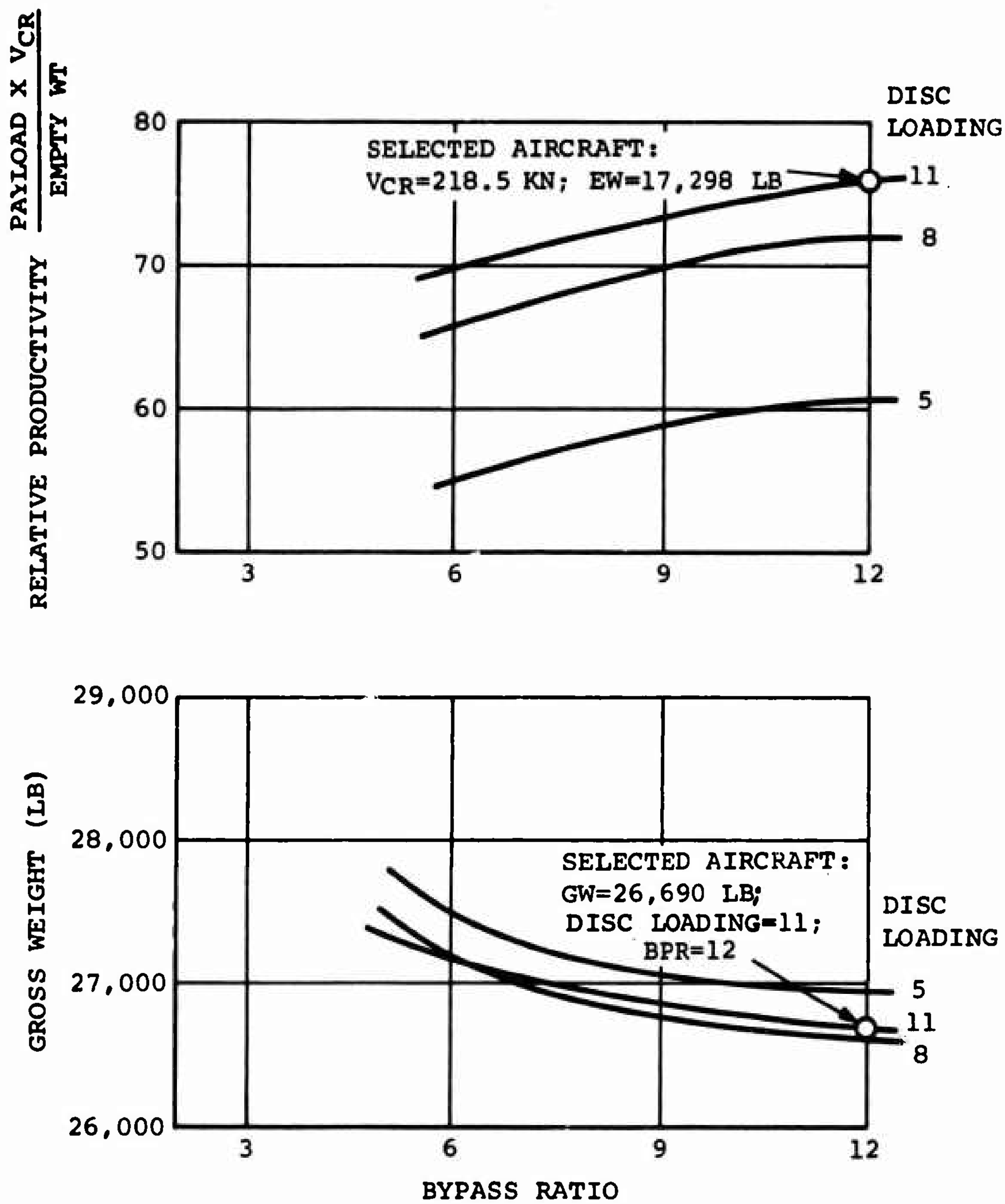


Figure 91. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Lift/Propulsion-Unloaded Aircraft Propulsion System 1a

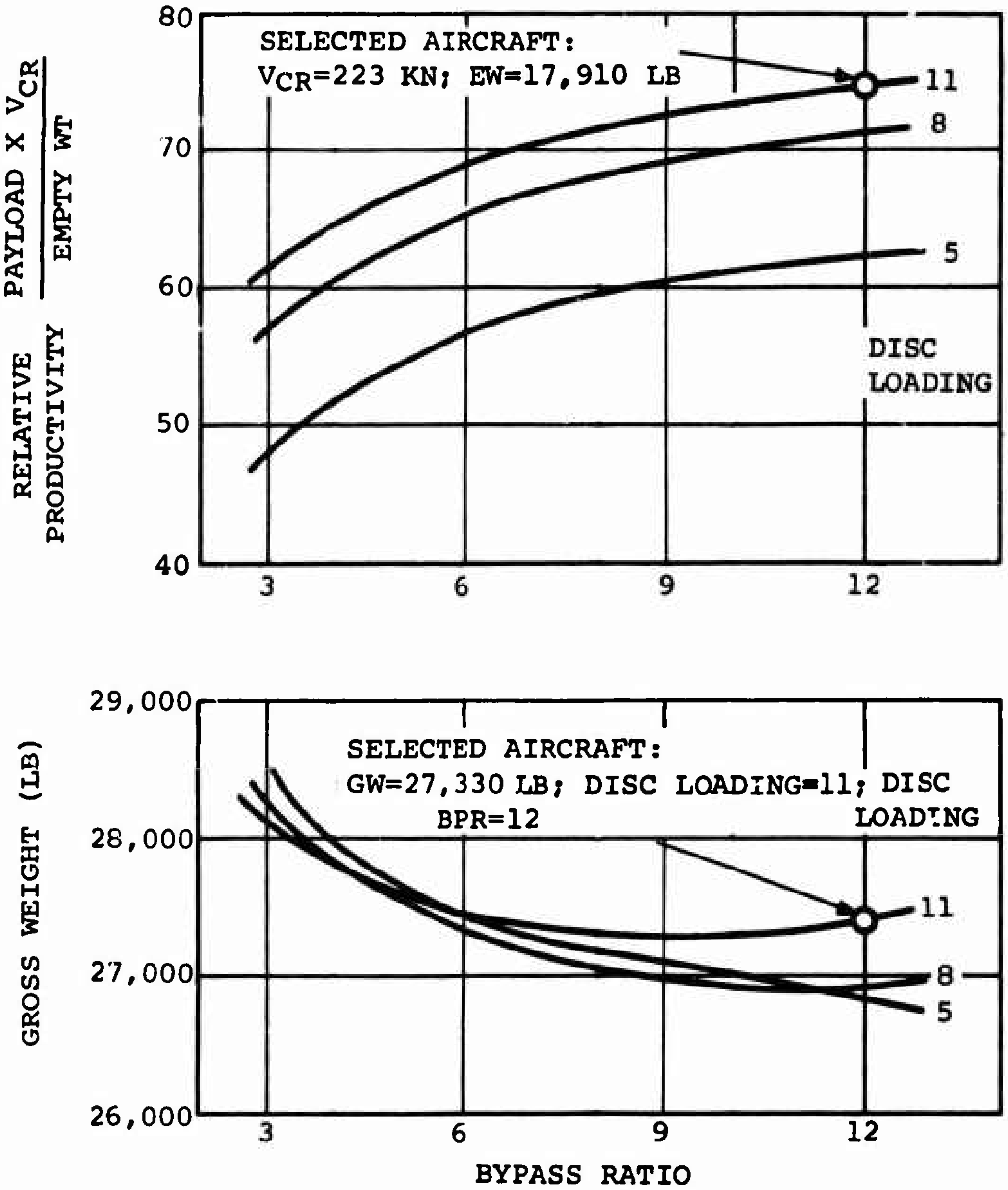


Figure 92. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Lift/Propulsion-Unloaded Aircraft Propulsion System 1b

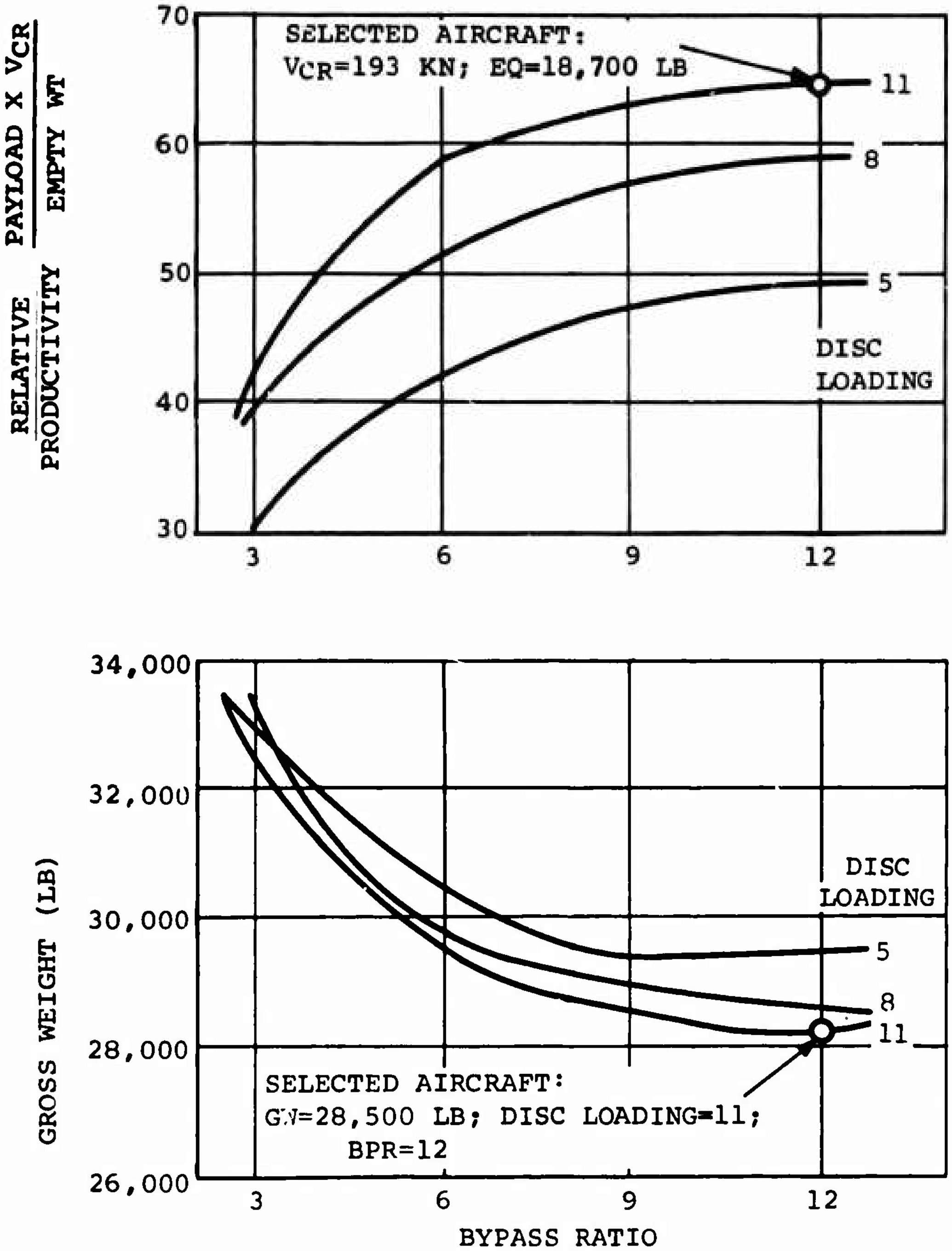


Figure 93. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2a

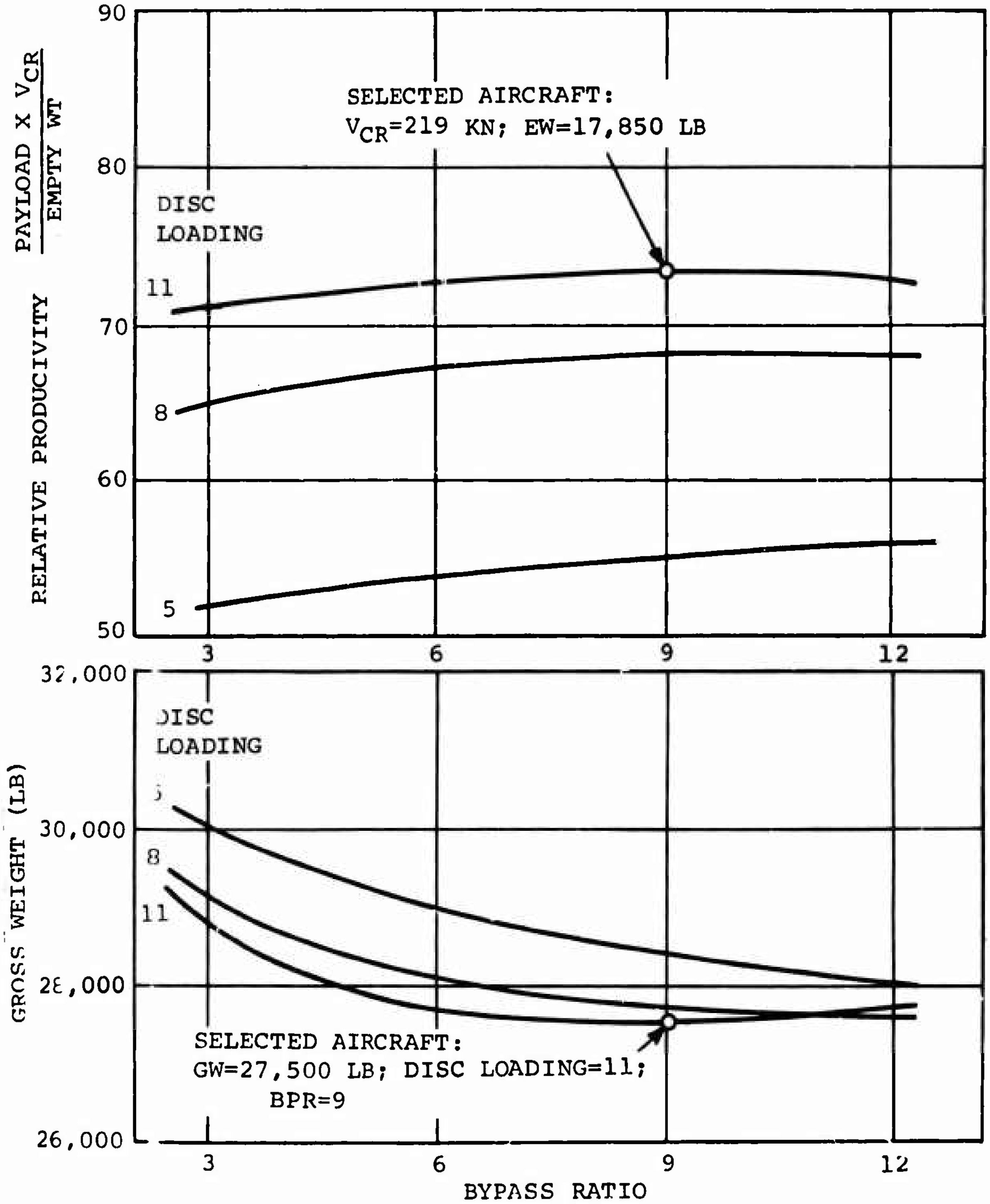


Figure 94 . Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Lift/Propulsion-Unloaded Aircraft Propulsion System 2b

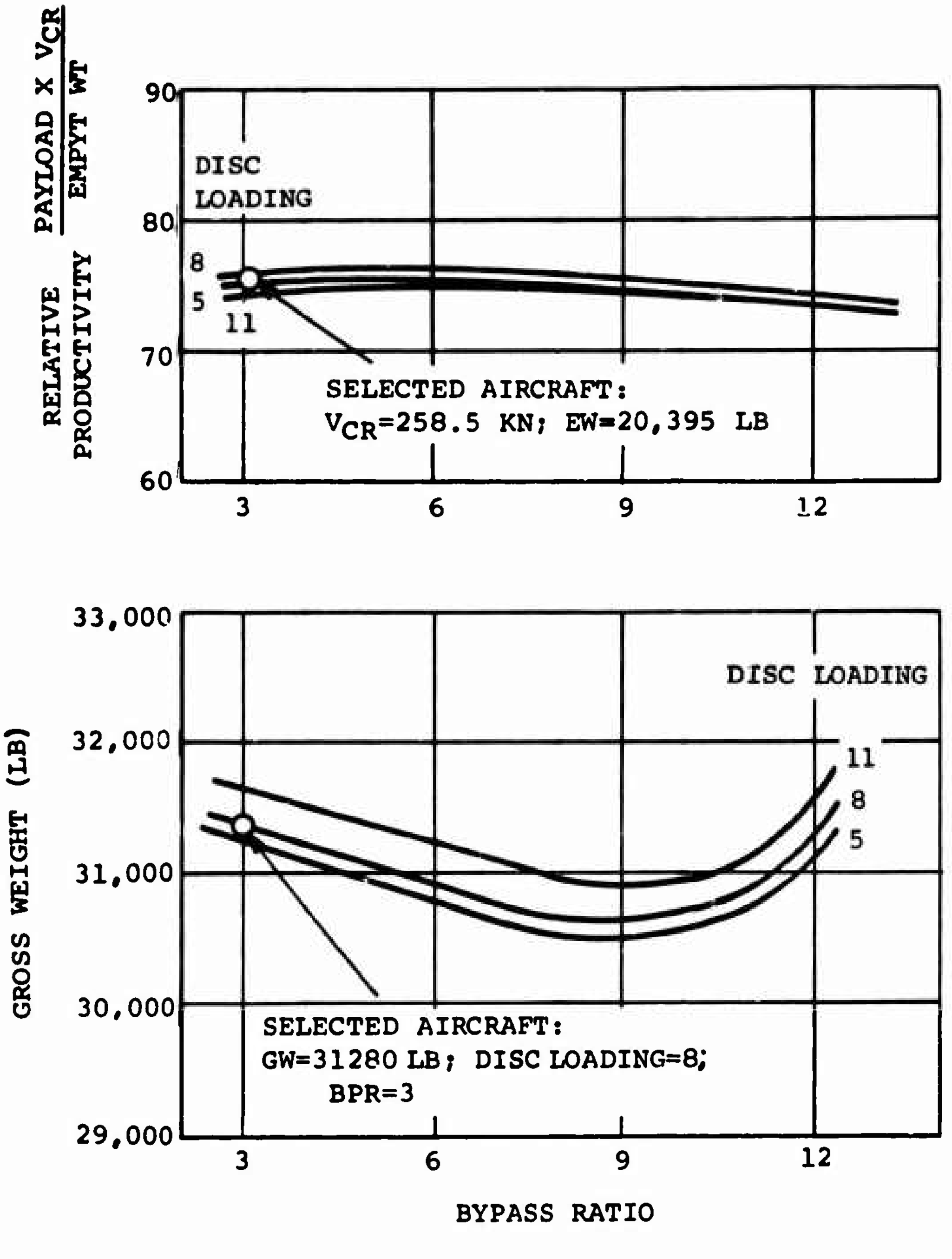


Figure 95. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio – Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 3

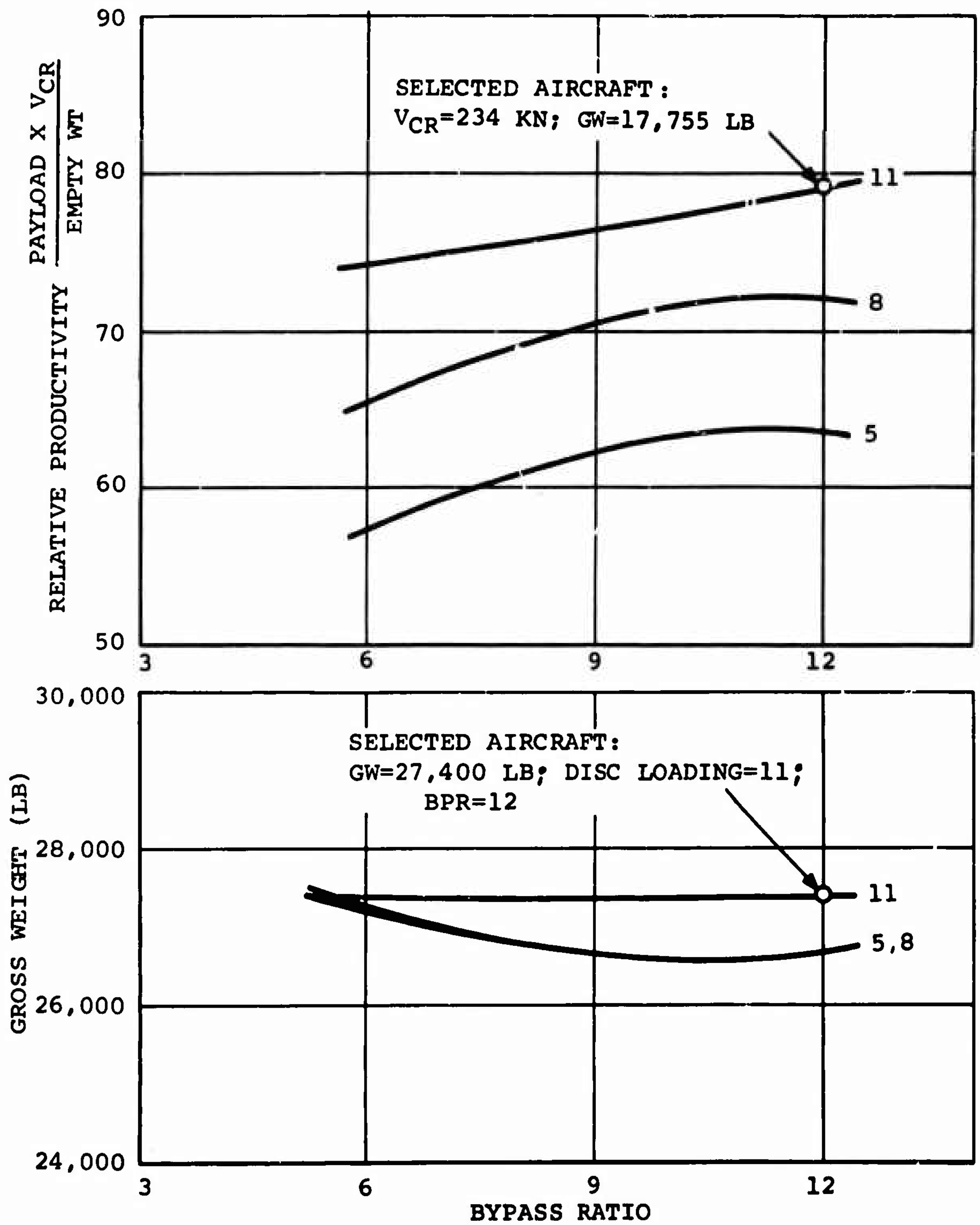


Figure 96. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Lift/Propulsion-Unloaded Aircraft Propulsion System 1a

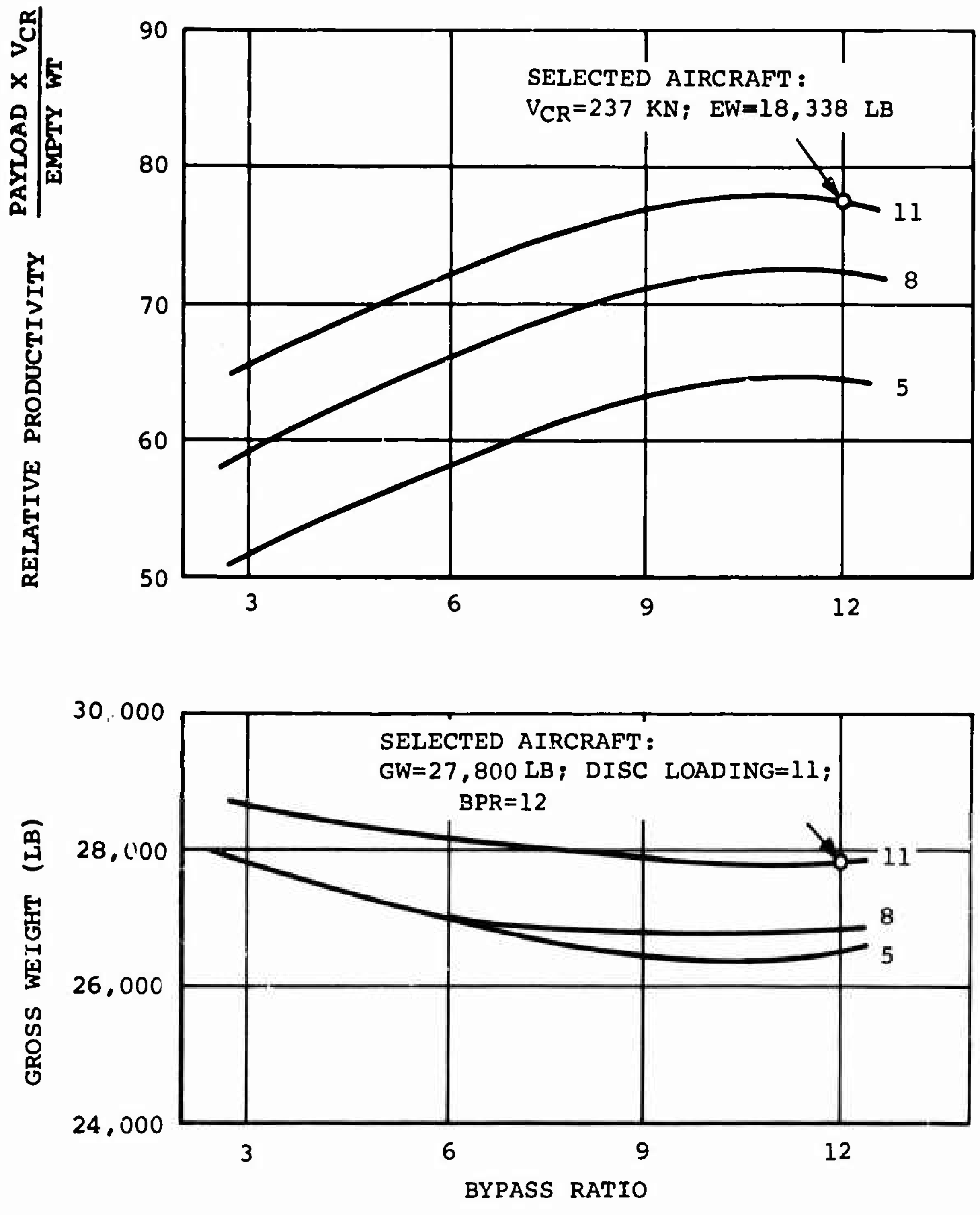


Figure 97. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 1b

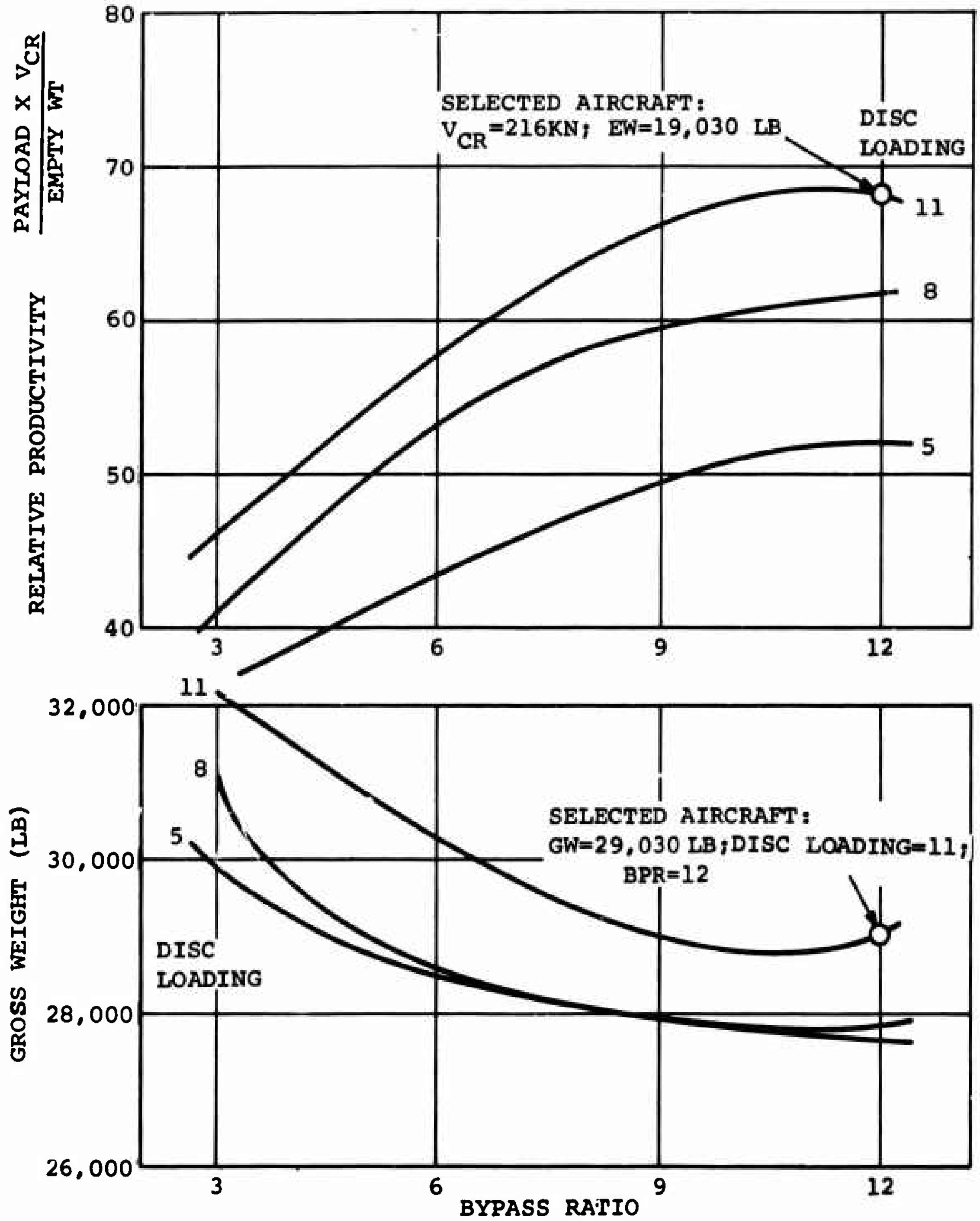


Figure 98. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2a

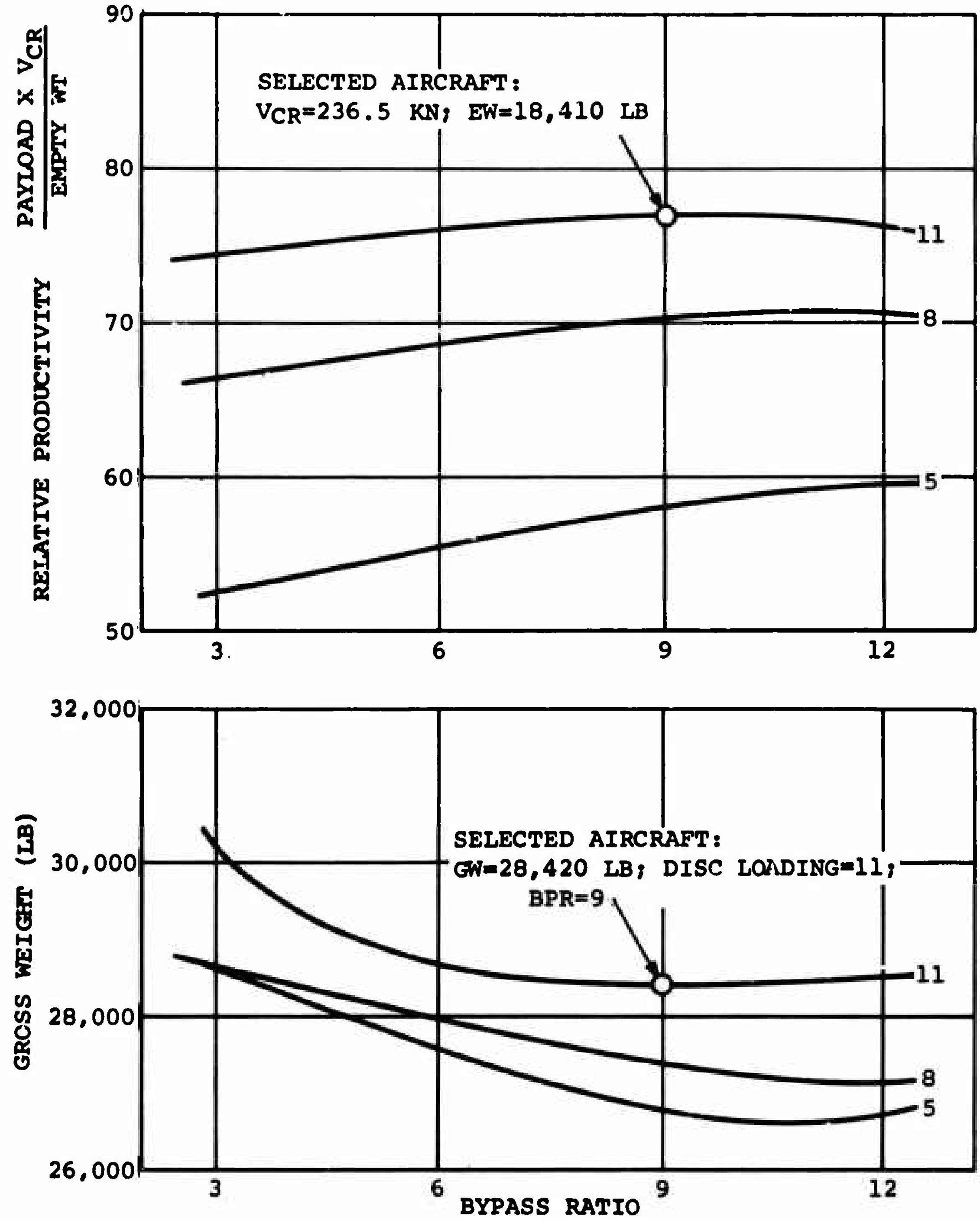


Figure 99. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2b

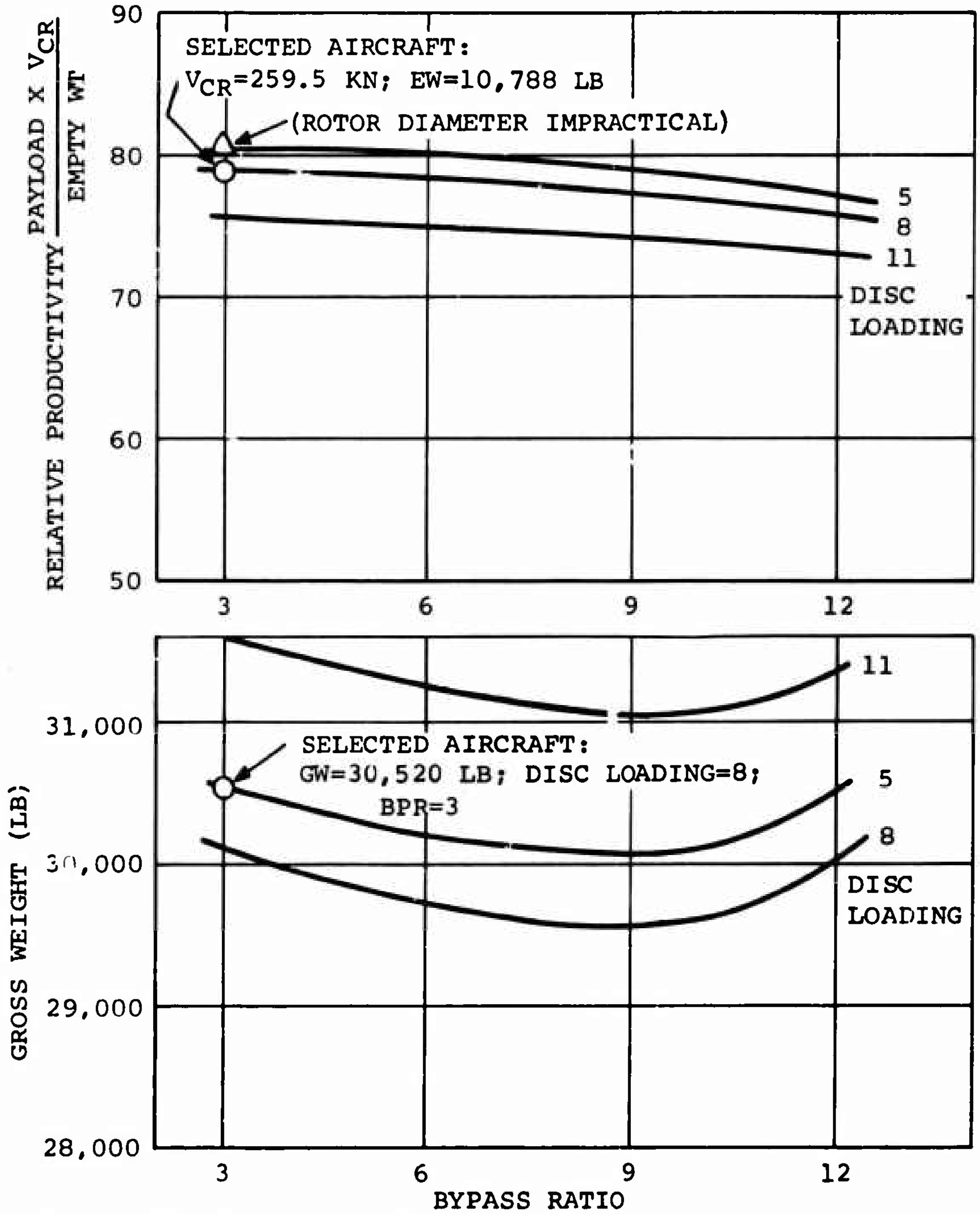


Figure 100. Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 3

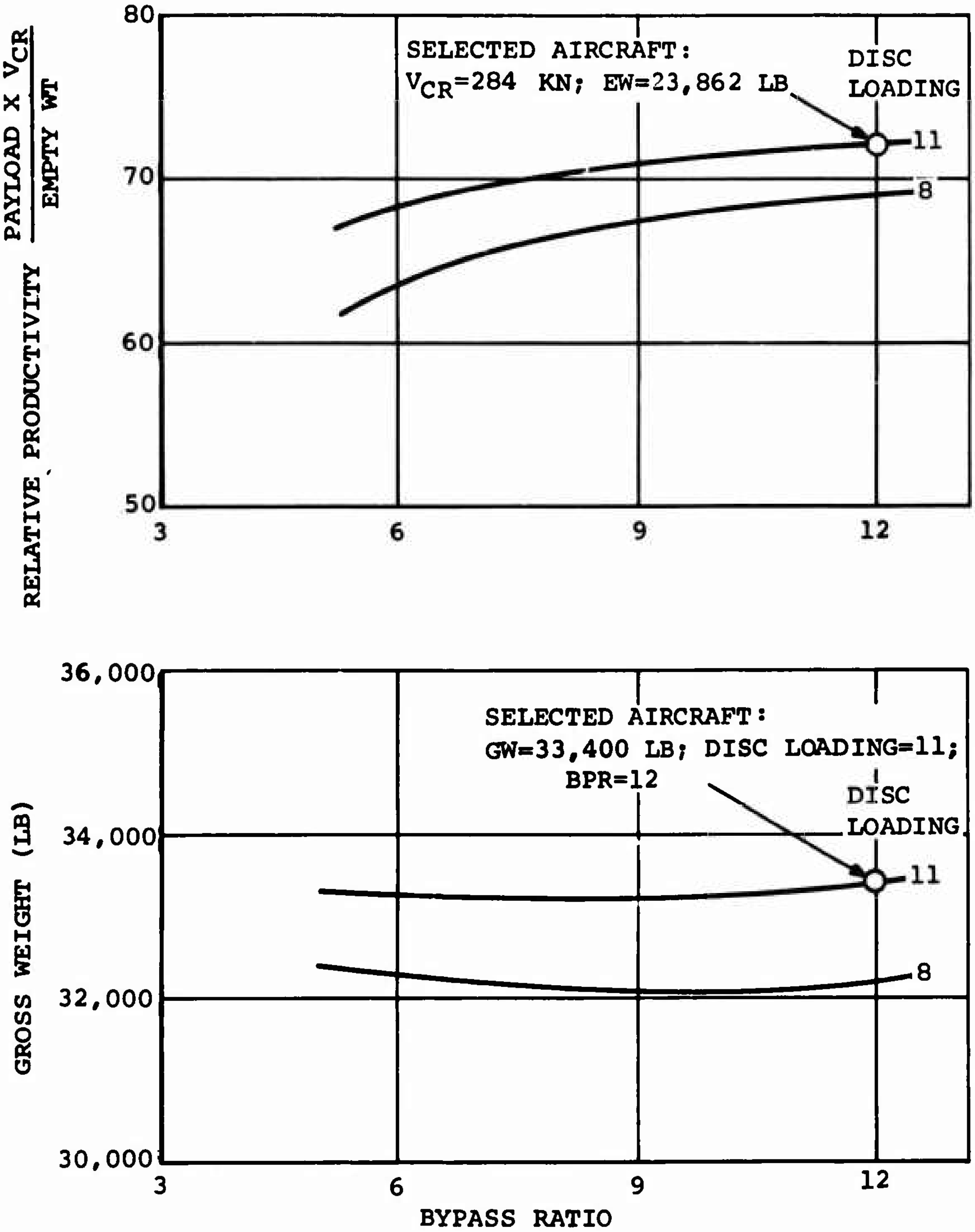


Figure 101. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Composite Aircraft Propulsion System 1a

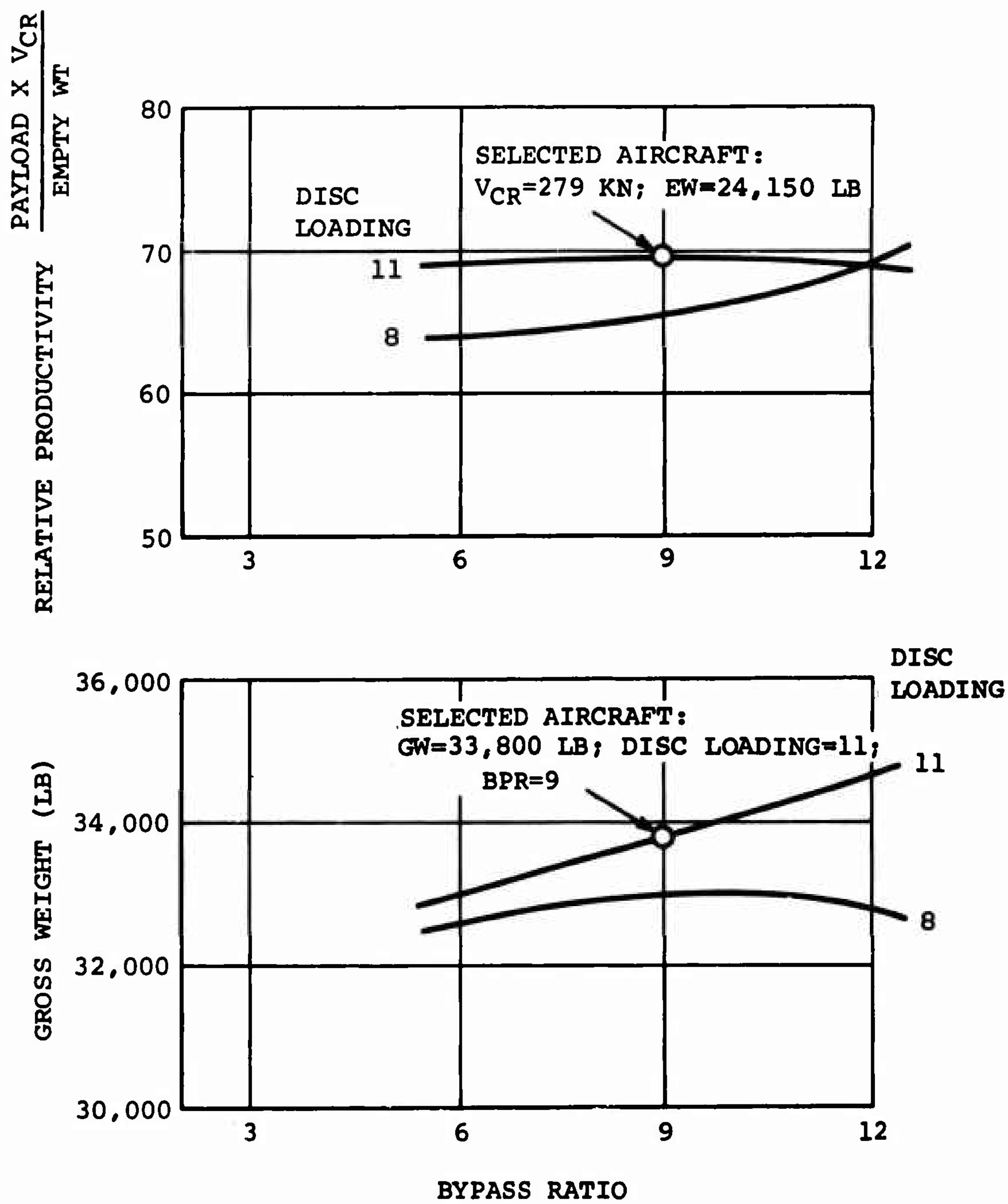


Figure 102. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Composite Aircraft Propulsion System 1b

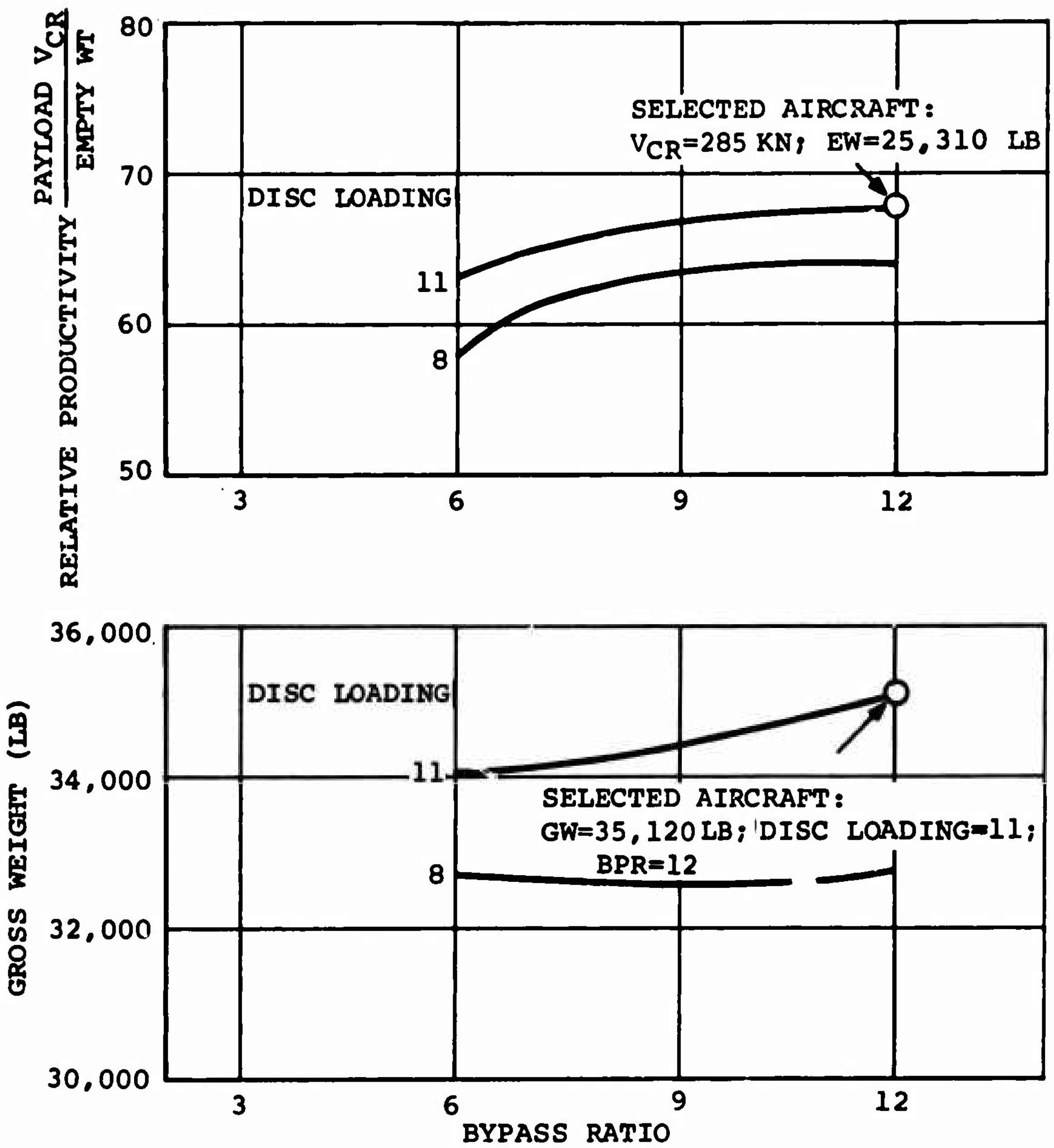
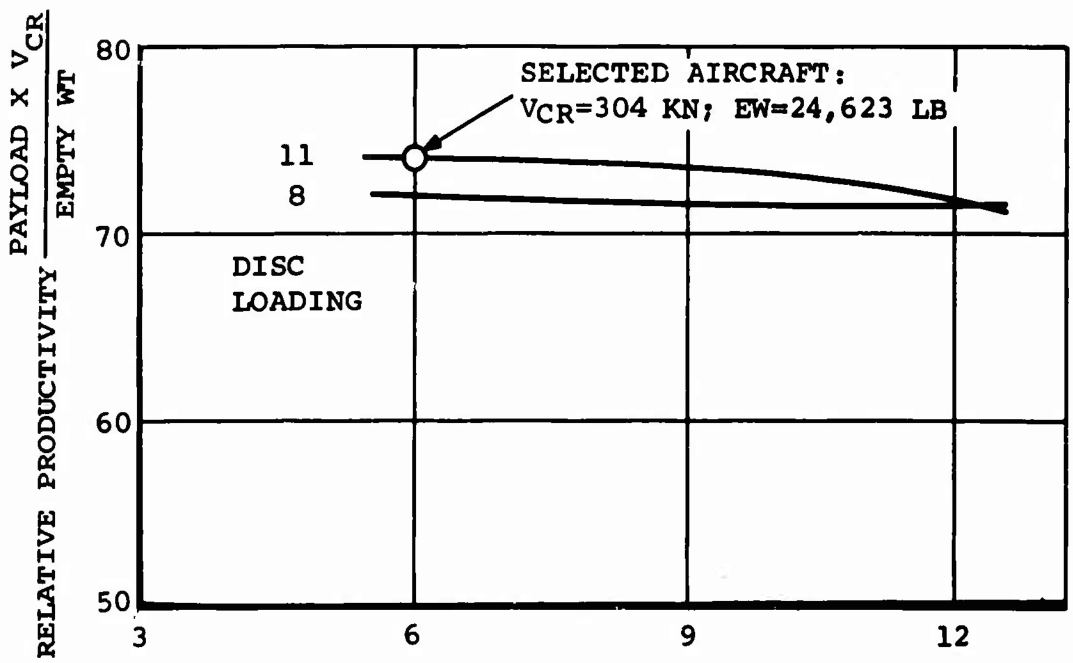


Figure 103. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Composite Aircraft Propulsion System 2a



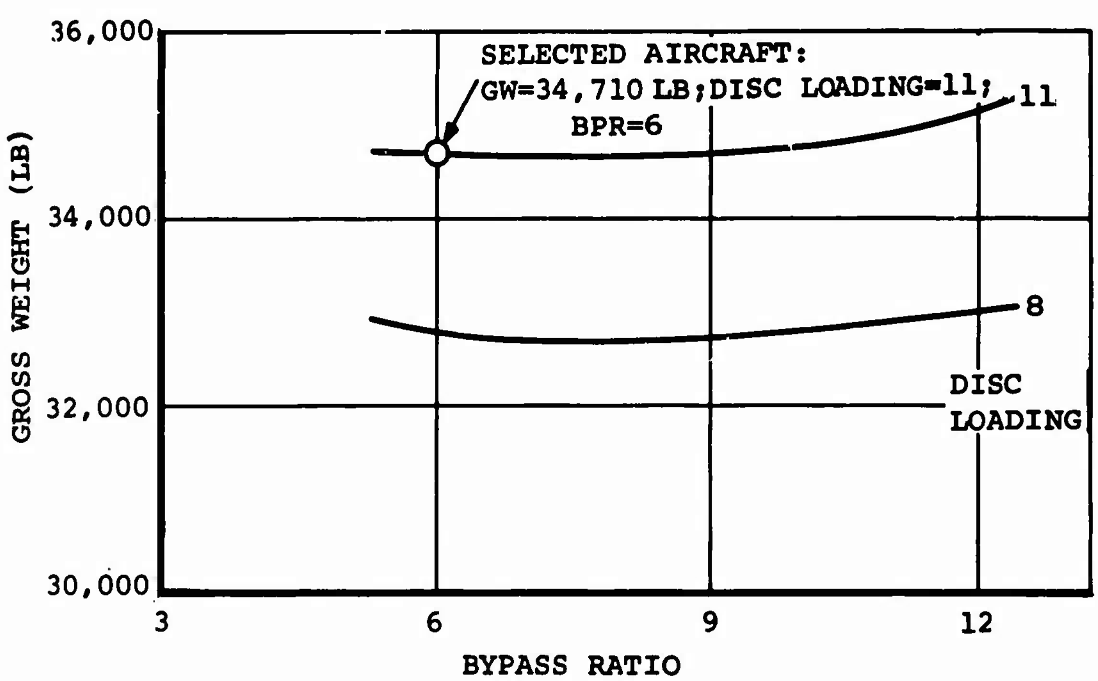


Figure 104 Variation of Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Composite Aircraft Propulsion System 2b

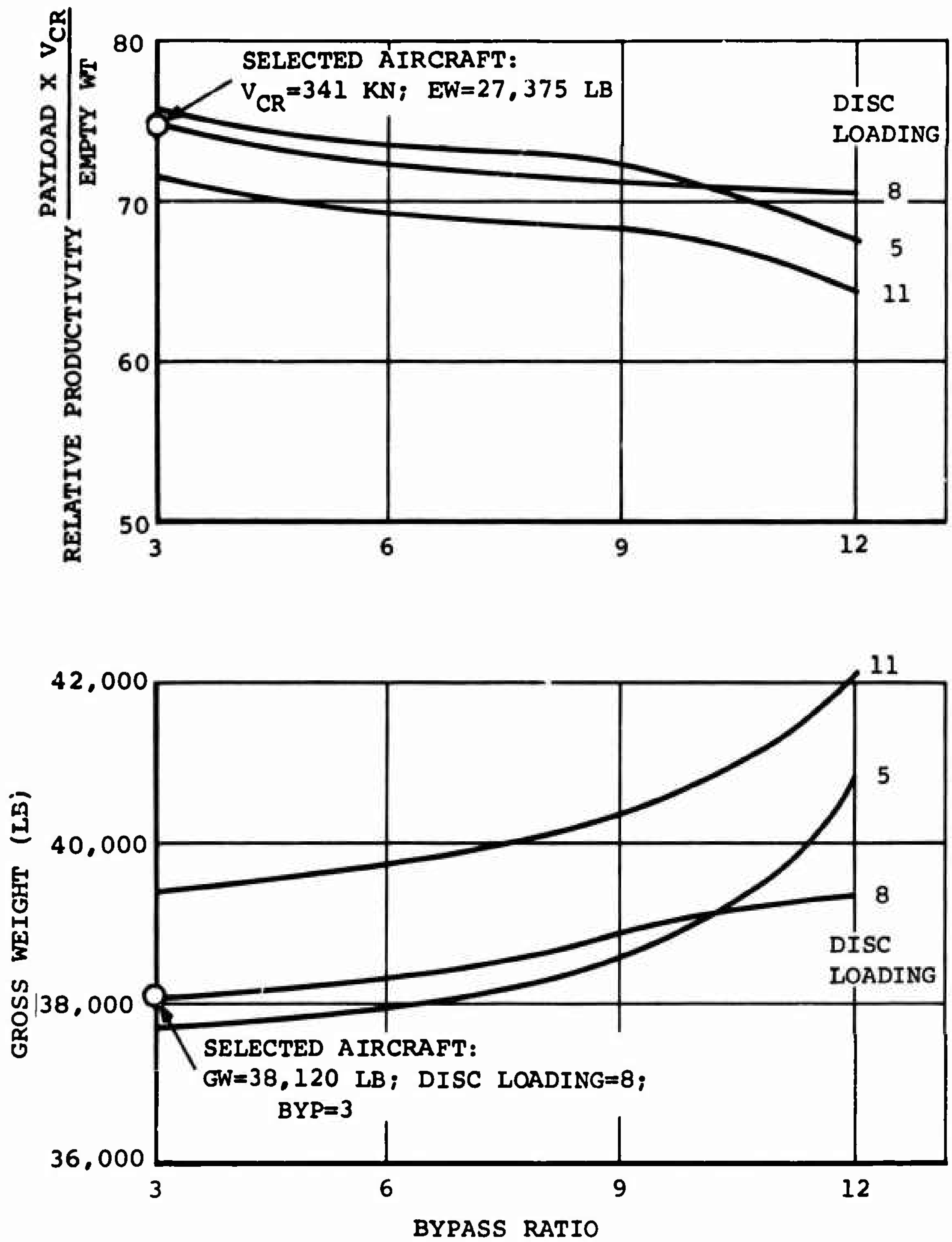


Figure 105. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Tandem Rotor Composite Aircraft Propulsion System 3

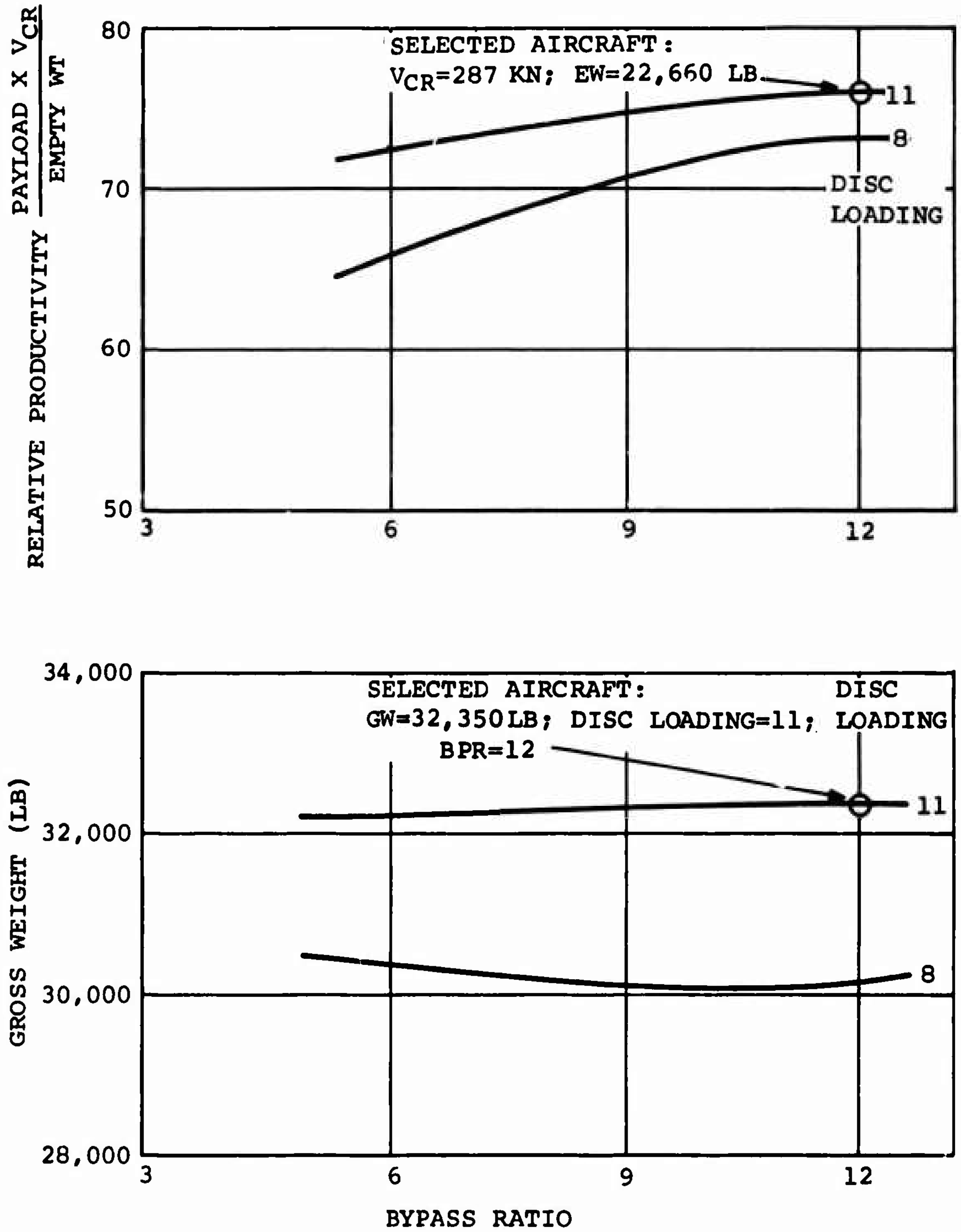


Figure 106. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Composite Aircraft Propulsion System 1a

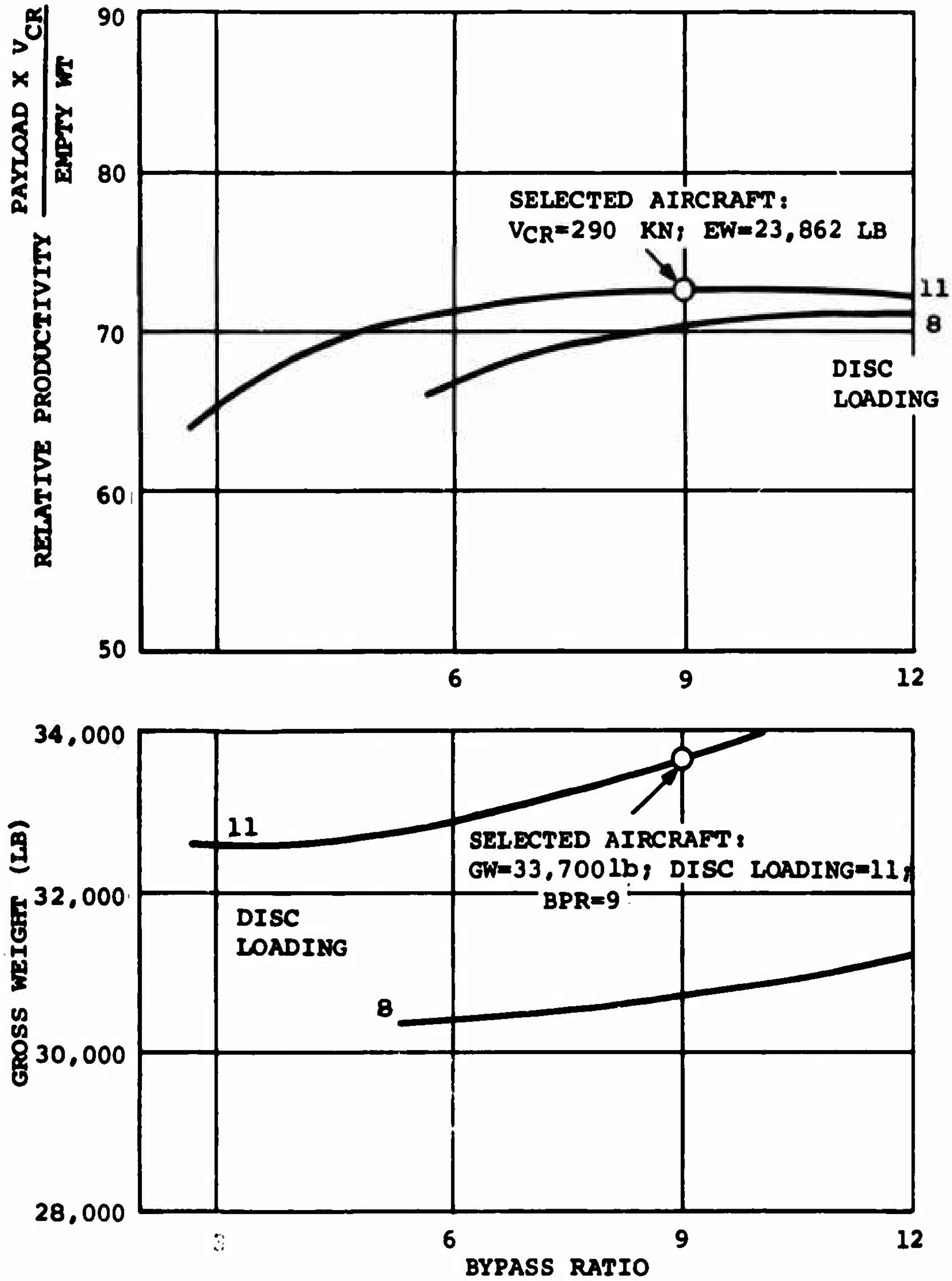


Figure 107. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Composite Aircraft Propulsion System 1b

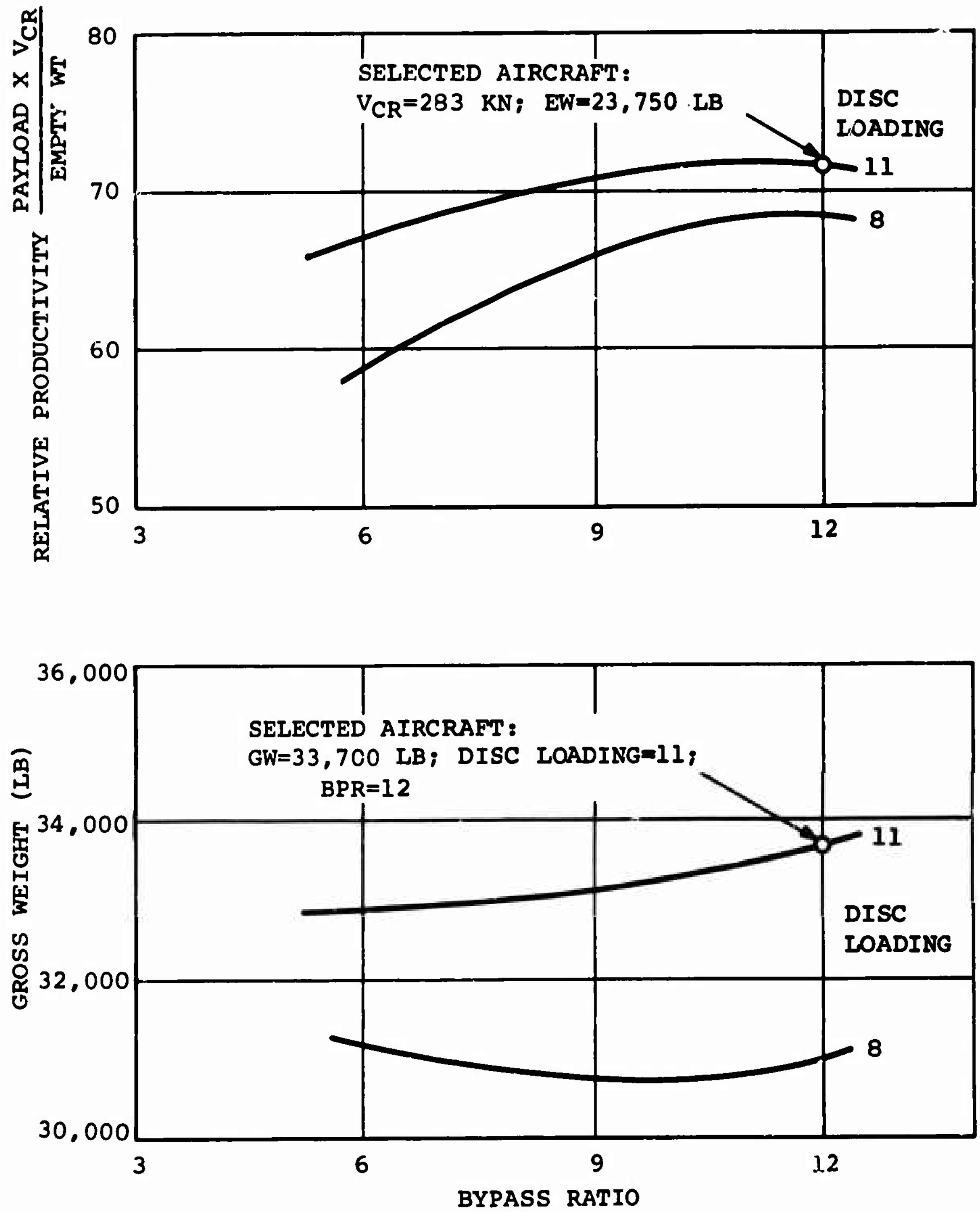


Figure 108. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Composite Aircraft Propulsion System 2a

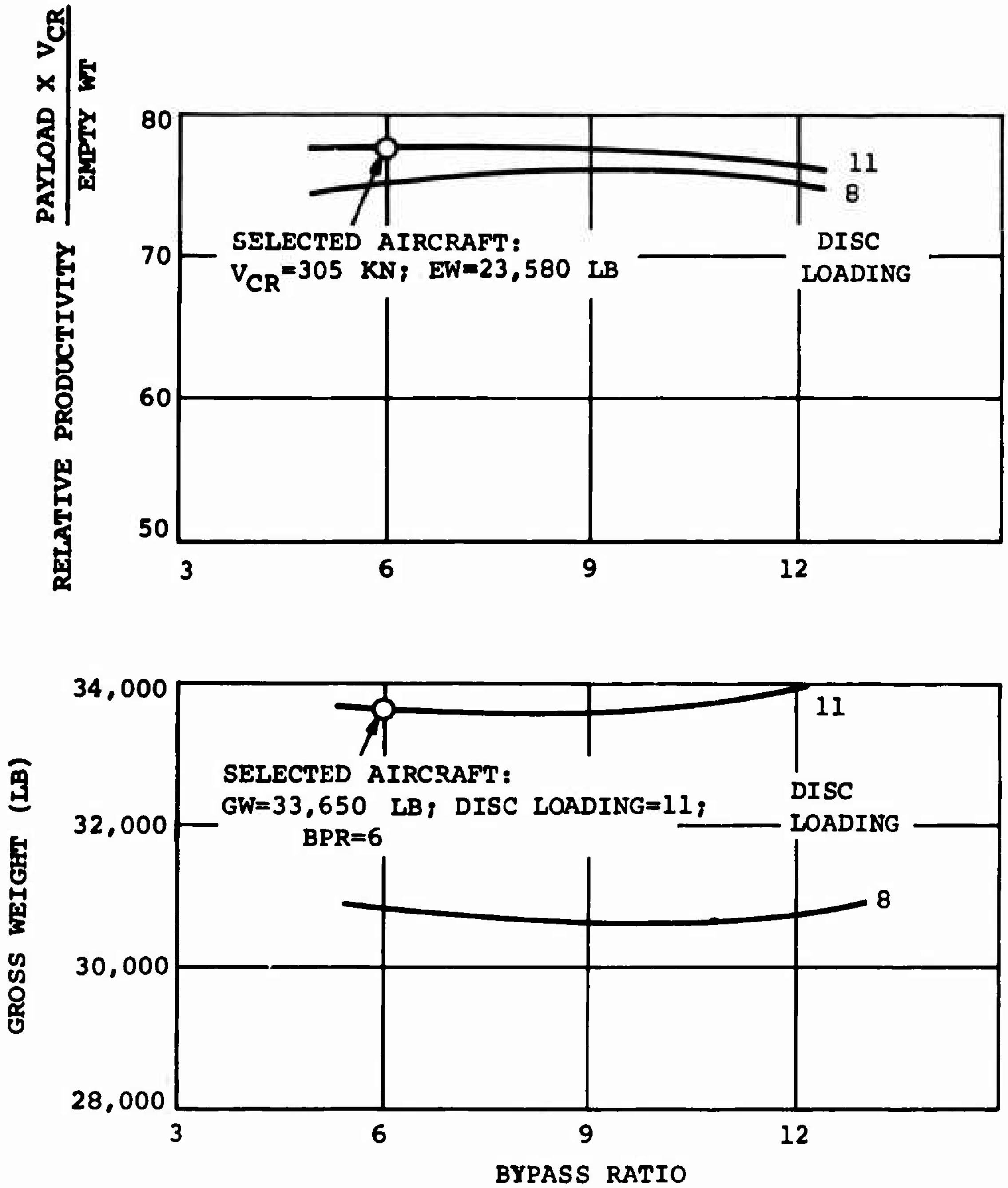


Figure 109. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Composite Aircraft Propulsion System 2b

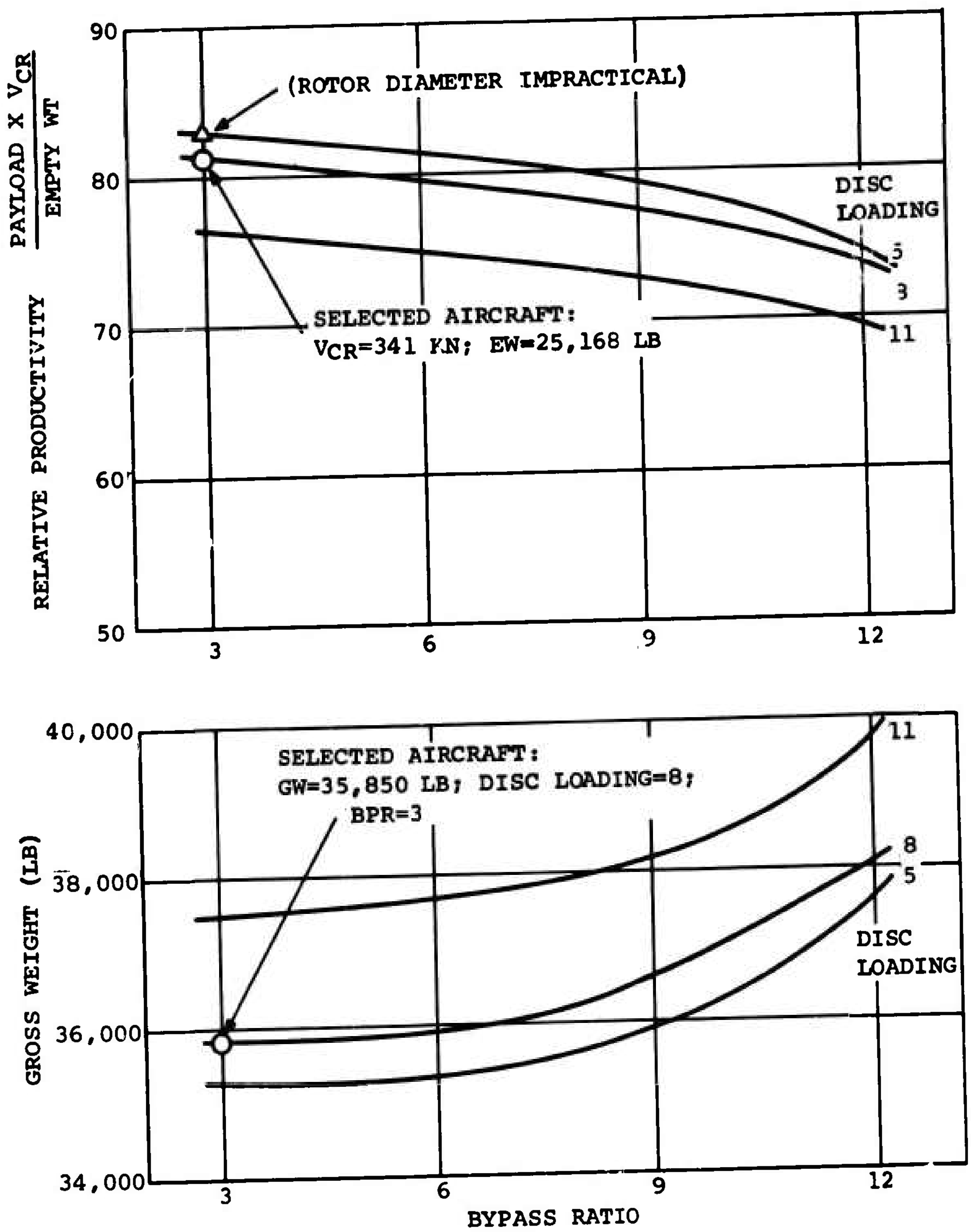


Figure 110. Variation of Design Mission Gross Weight and Relative Productivity With Disc Loading and Bypass Ratio - Single Rotor Composite Aircraft Propulsion System 3

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BASIC DATA

1. TWO ENGINES - EACH 2865 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 4220-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 253 SQ FT
5. DESIGN GROSS WEIGHT - 26,690 LB (LF=3.0)
6. EMPTY WEIGHT - 17,298 LB

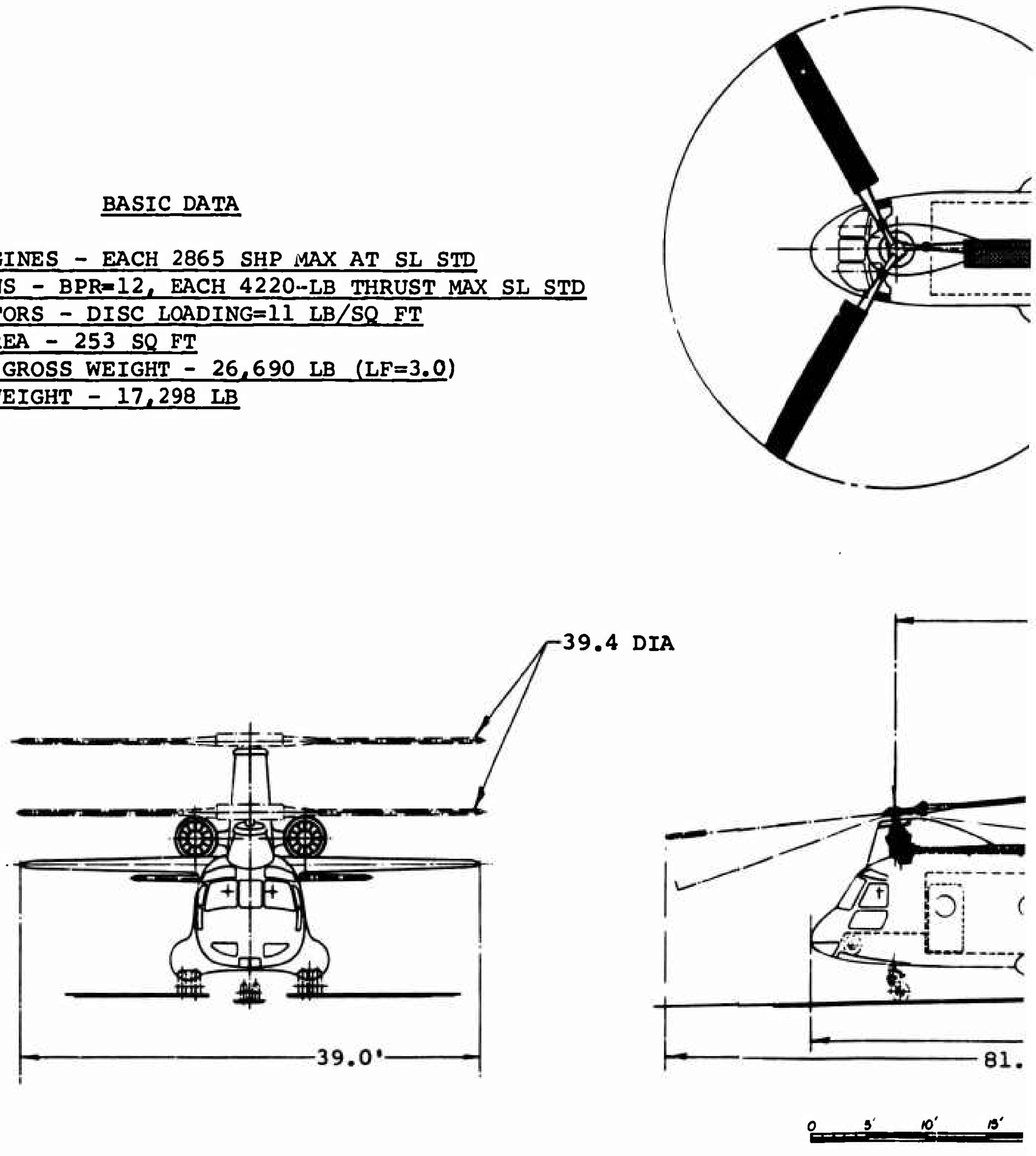
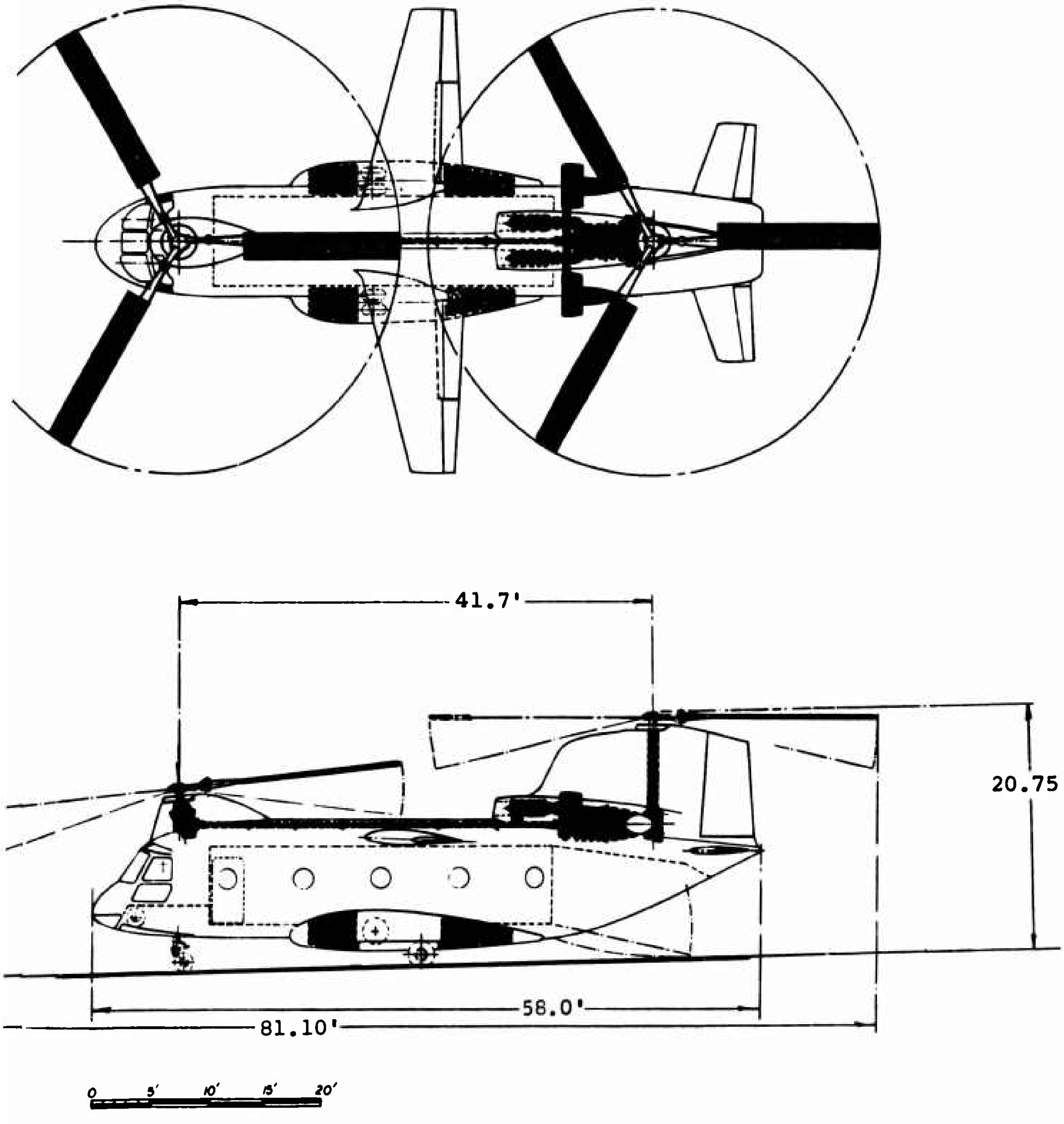


Figure 111. Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 1a

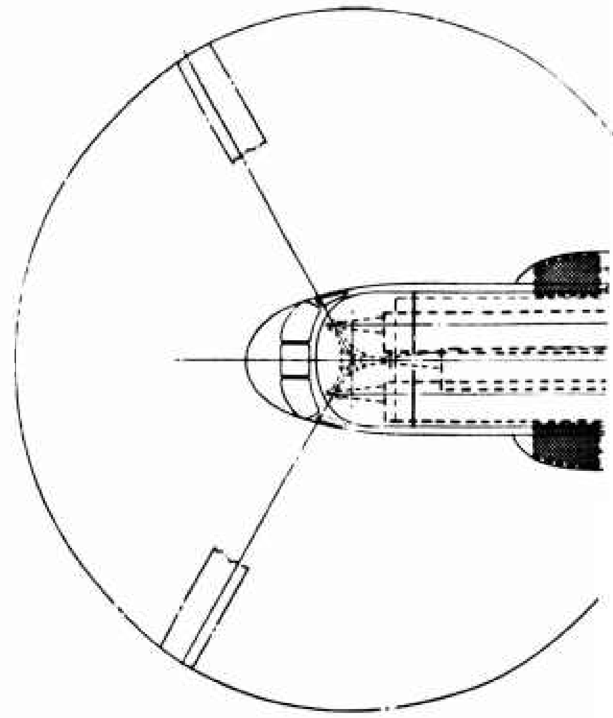
A

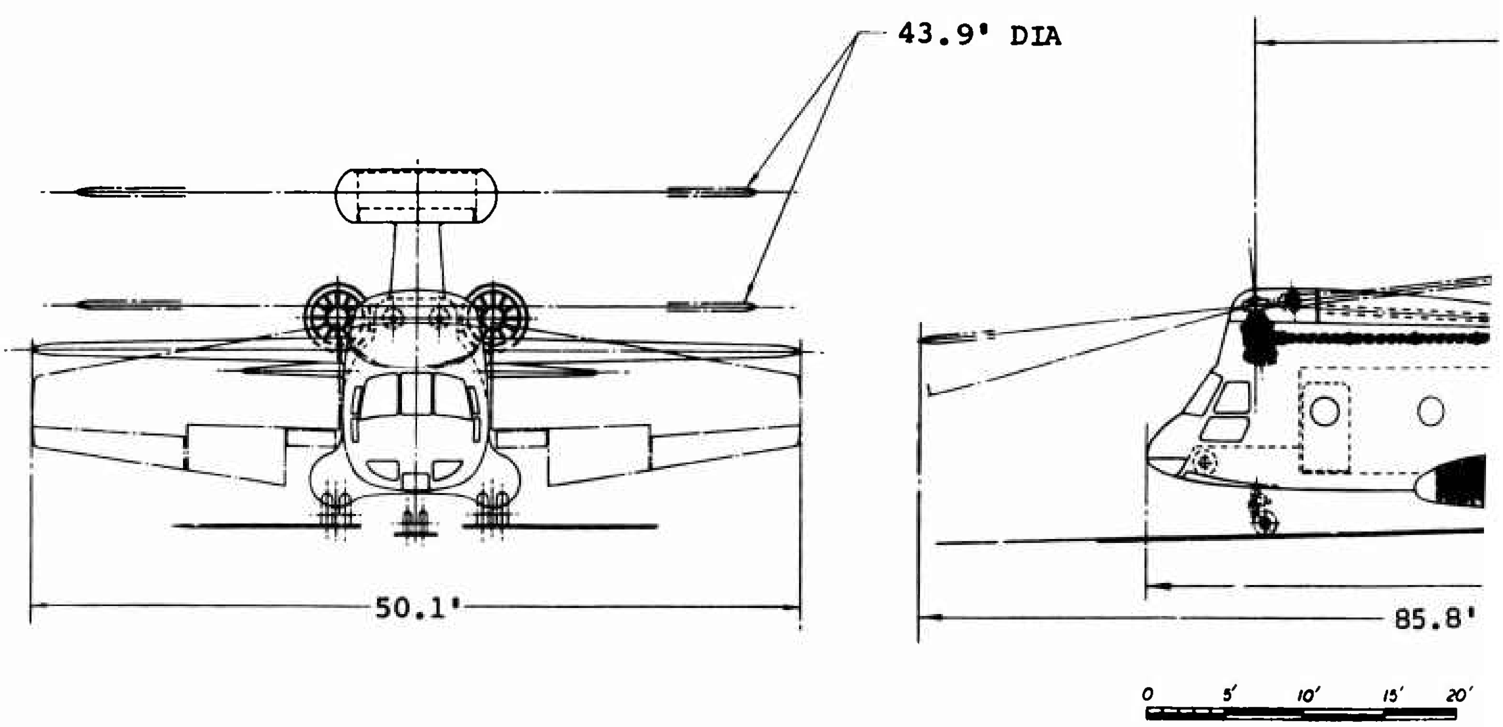


41.7'
20.75'
58.0'
81.10'
0
5'
10'
15'
20'

BASIC DATA

1. TWO ENGINES - EACH 3950 SHP MAX AT SL STD
2. TWO FANS BPR=12, EACH 5800-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 418 SQ FT
5. DESIGN GROSS WEIGHT - 33,400 LB (LF=3.0)
6. EMPTY WEIGHT - 23,655 LB





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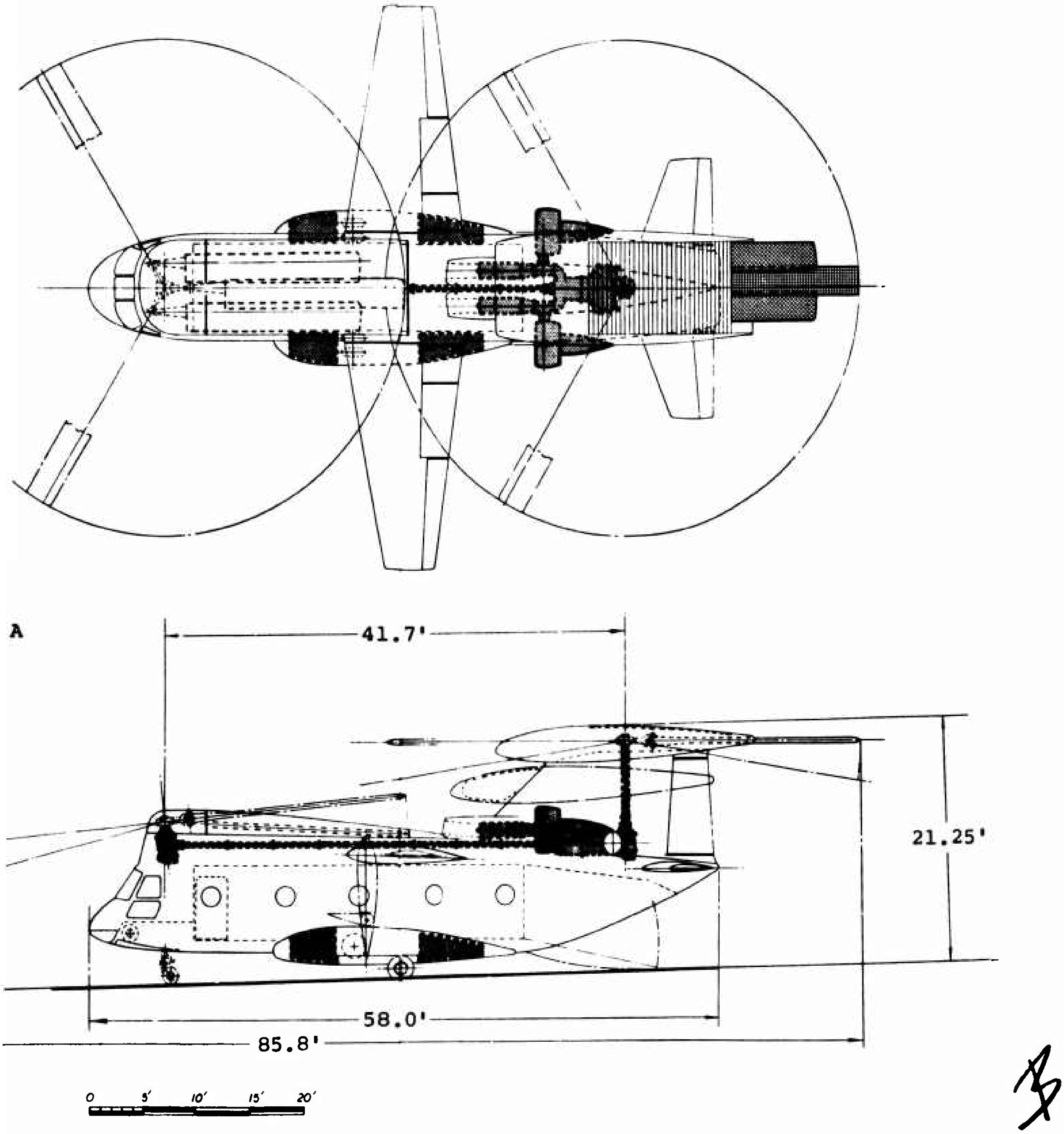
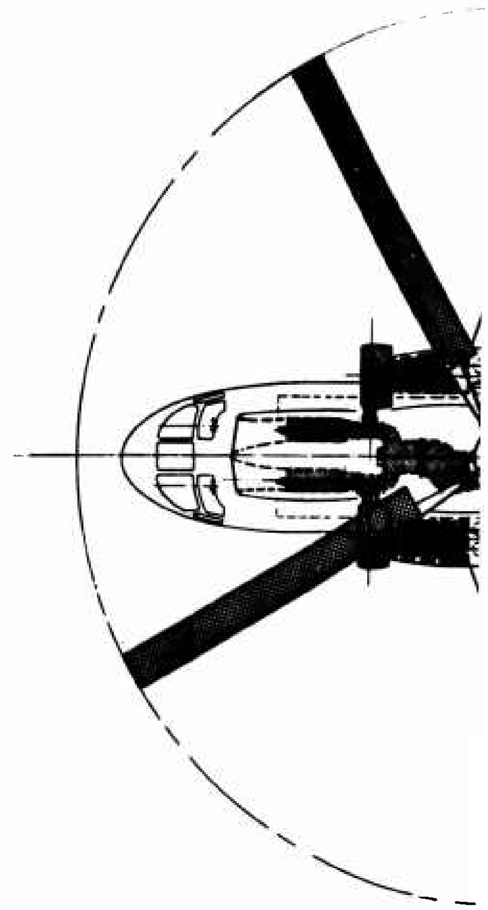
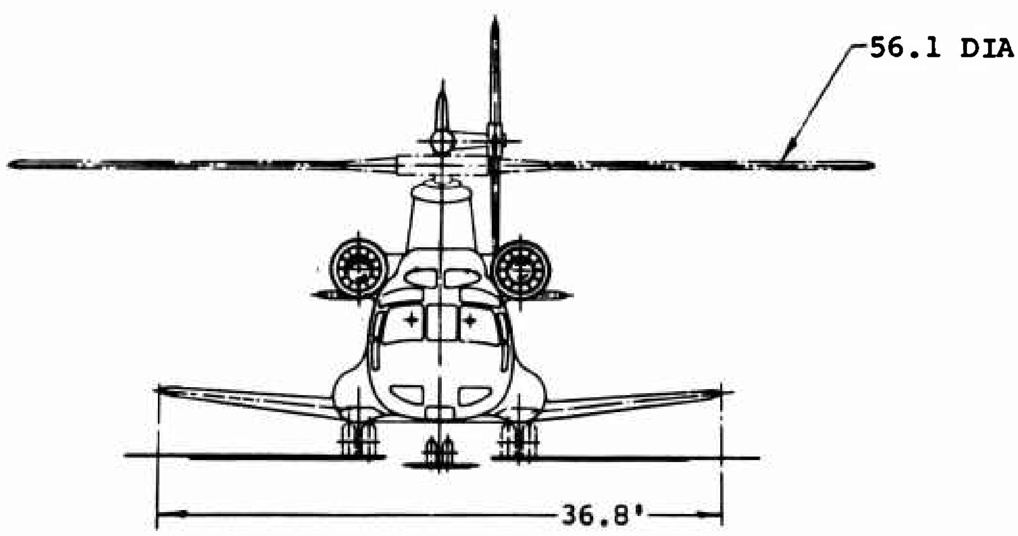


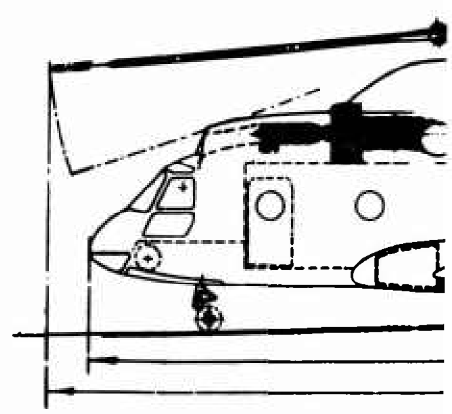
Figure 112. Tandem Rotor Composite Aircraft Propulsion System 1a



BASIC DATA

1. TWO ENGINES - EACH 3350 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 4930-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 225 SQ FT
5. DESIGN GROSS WEIGHT - 27,400 LB (LF=3.0)
6. EMPTY WEIGHT - 17,755 LB





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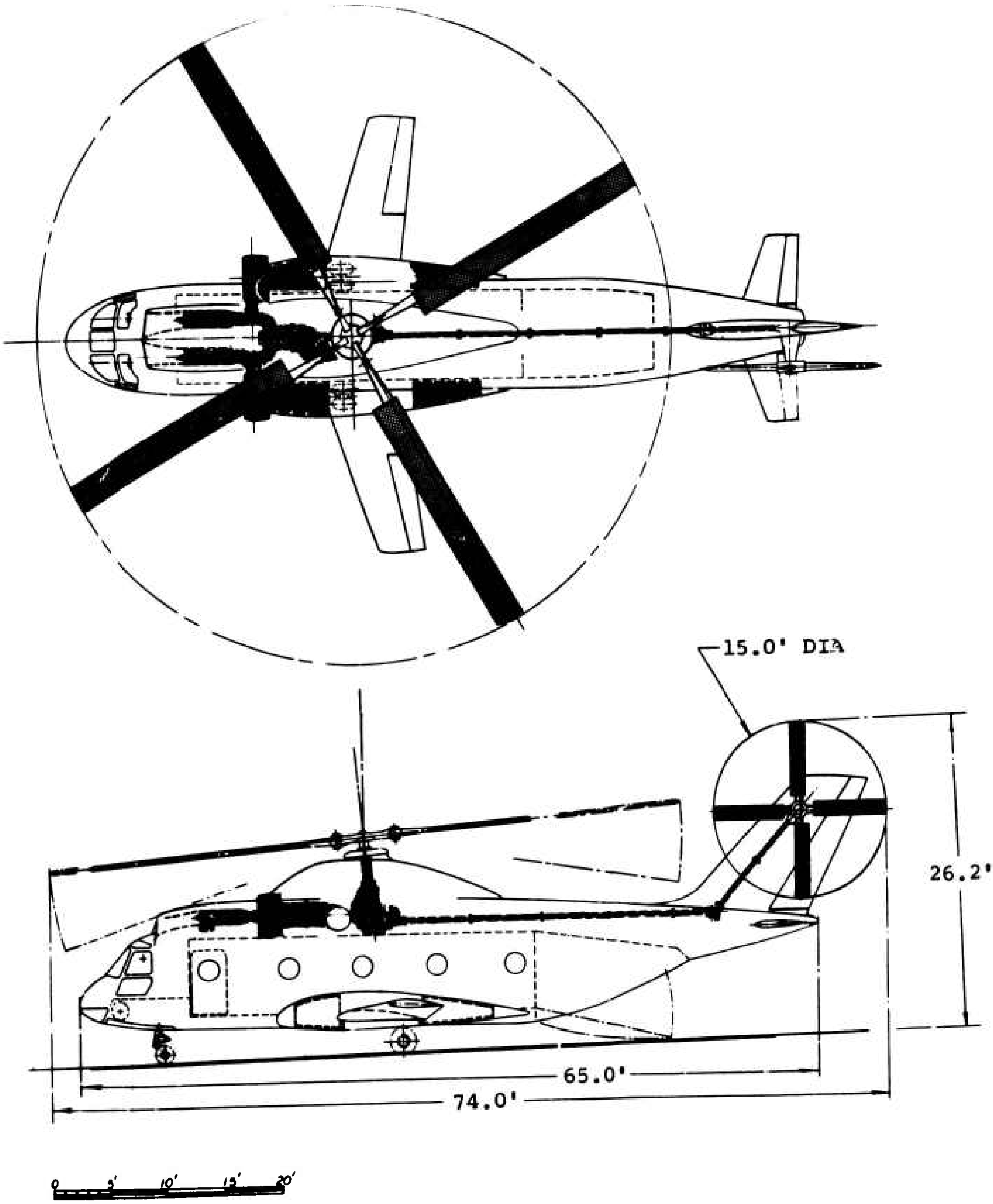
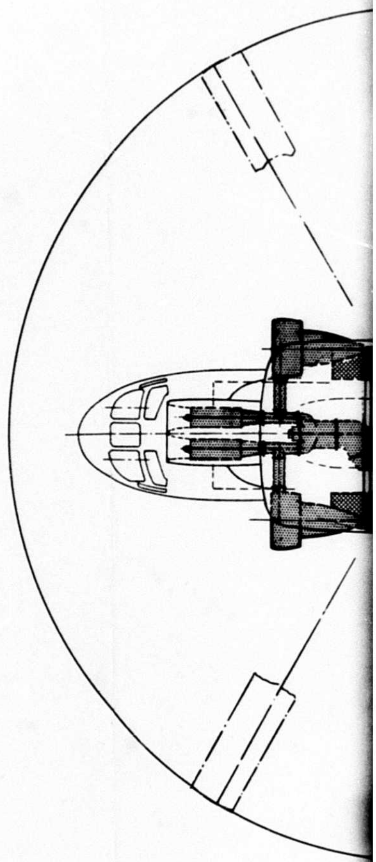
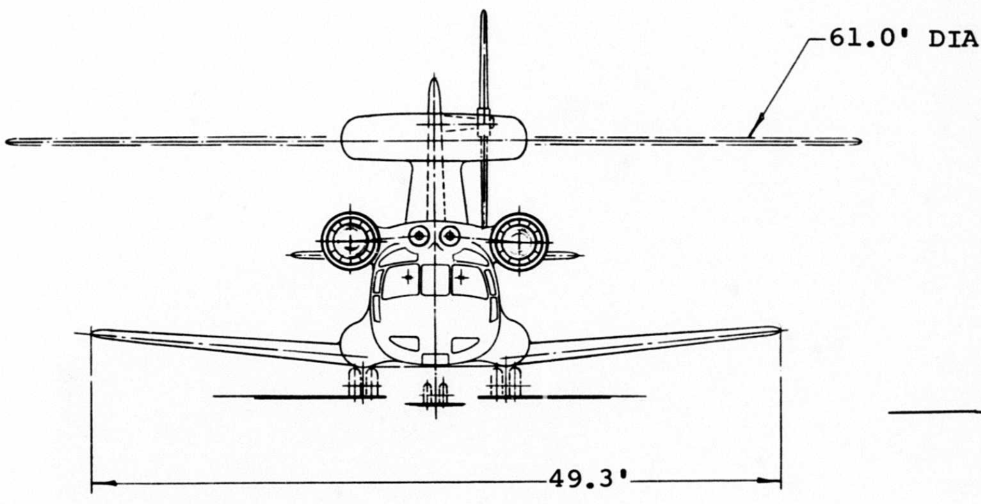


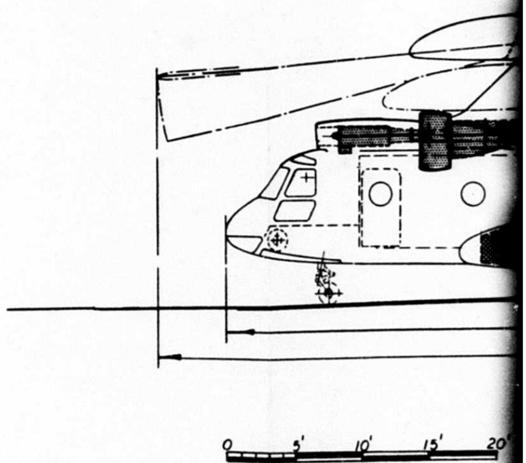
Figure 113. Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 1a



BASIC DATA

1. TWO ENGINES - EACH 4015 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 5880-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 405 SQ FT
5. DESIGN GROSS WEIGHT - 32,350 LB (LF=3.0)
6. EMPTY WEIGHT - 22,660 LB





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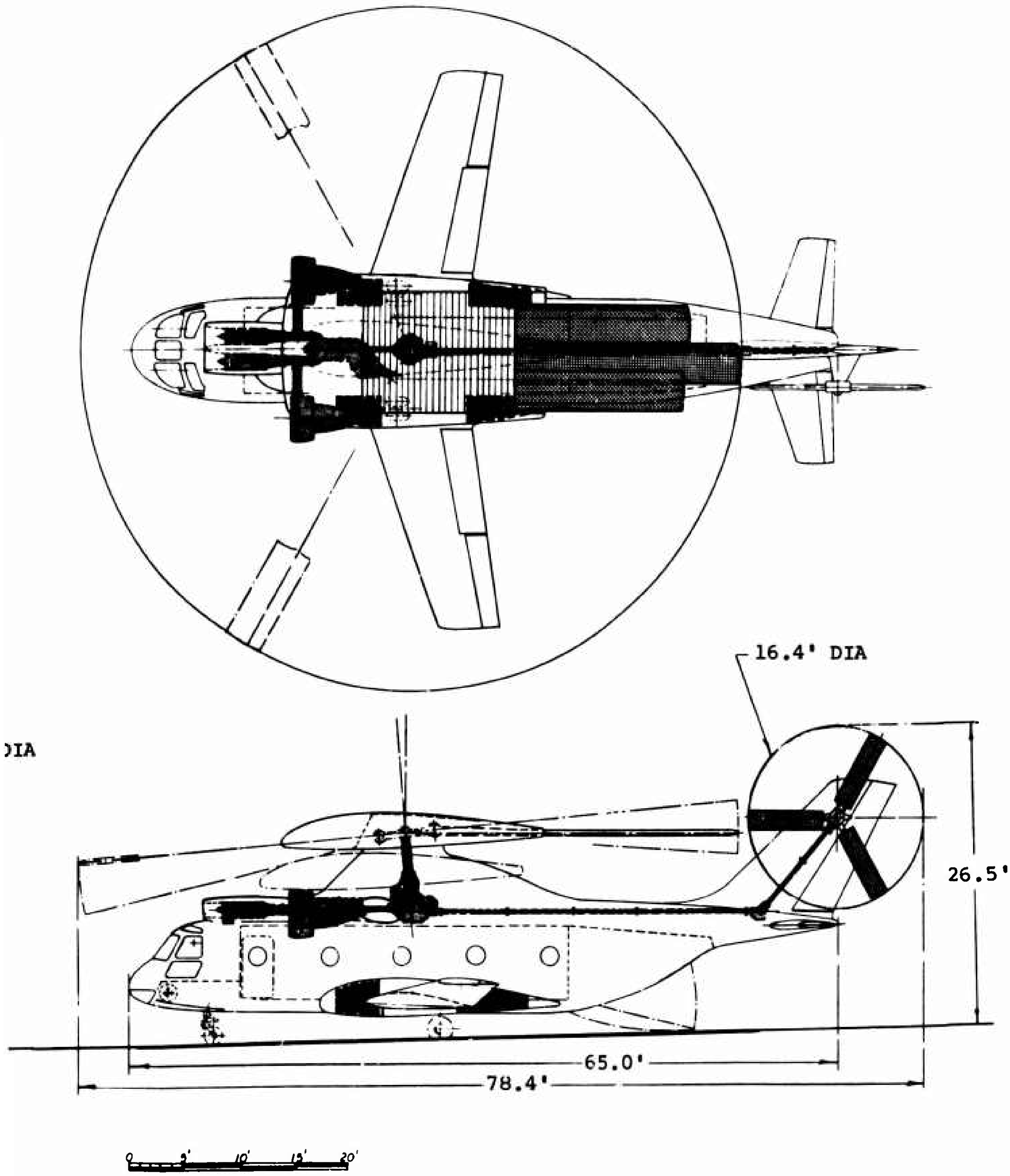

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Figure 114. Single Rotor Composite Aircraft Propulsion System 1a

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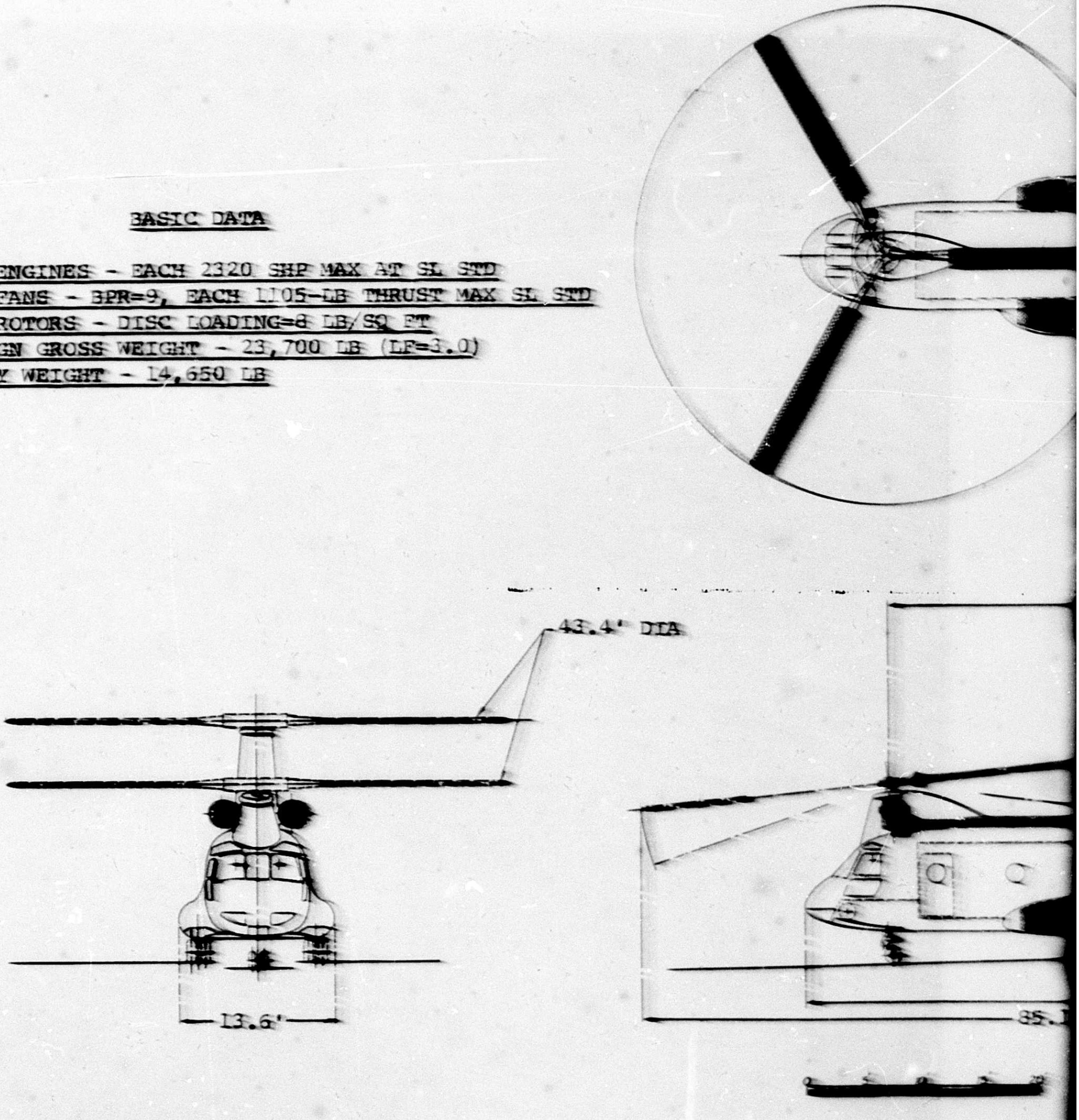
BASIC DATA

1. TWO ENGINES - EACH 2320 SHP MAX AT SL STD
2. TWO FANS - BPR=9, EACH 1105-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=8 LB/SQ FT
4. DESIGN GROSS WEIGHT - 23,700 LB (LF=3.0)
5. EMPTY WEIGHT - 14,650 LB



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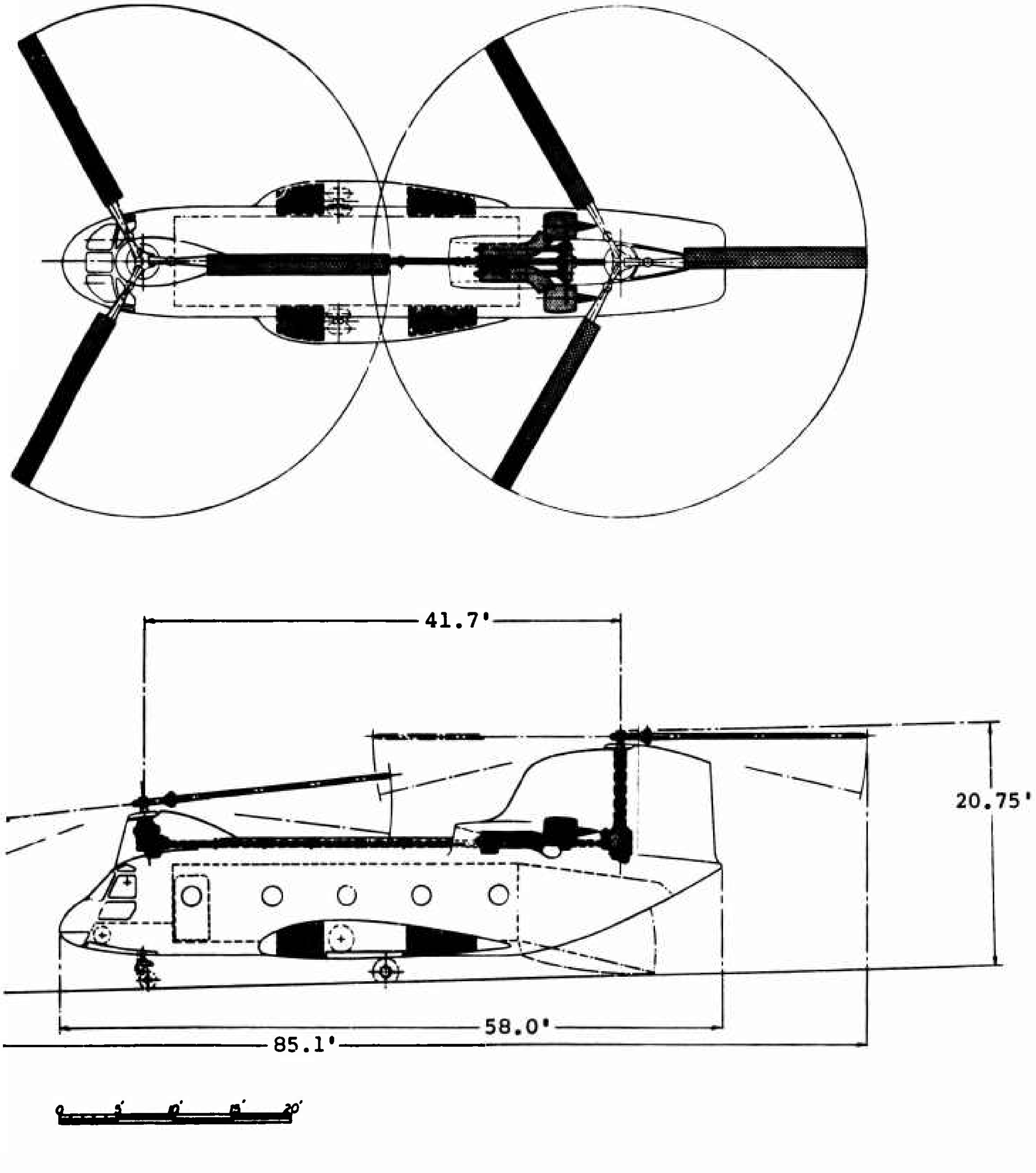
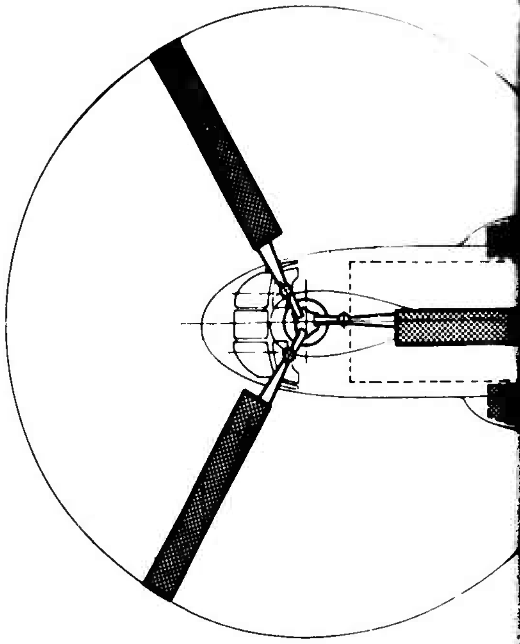
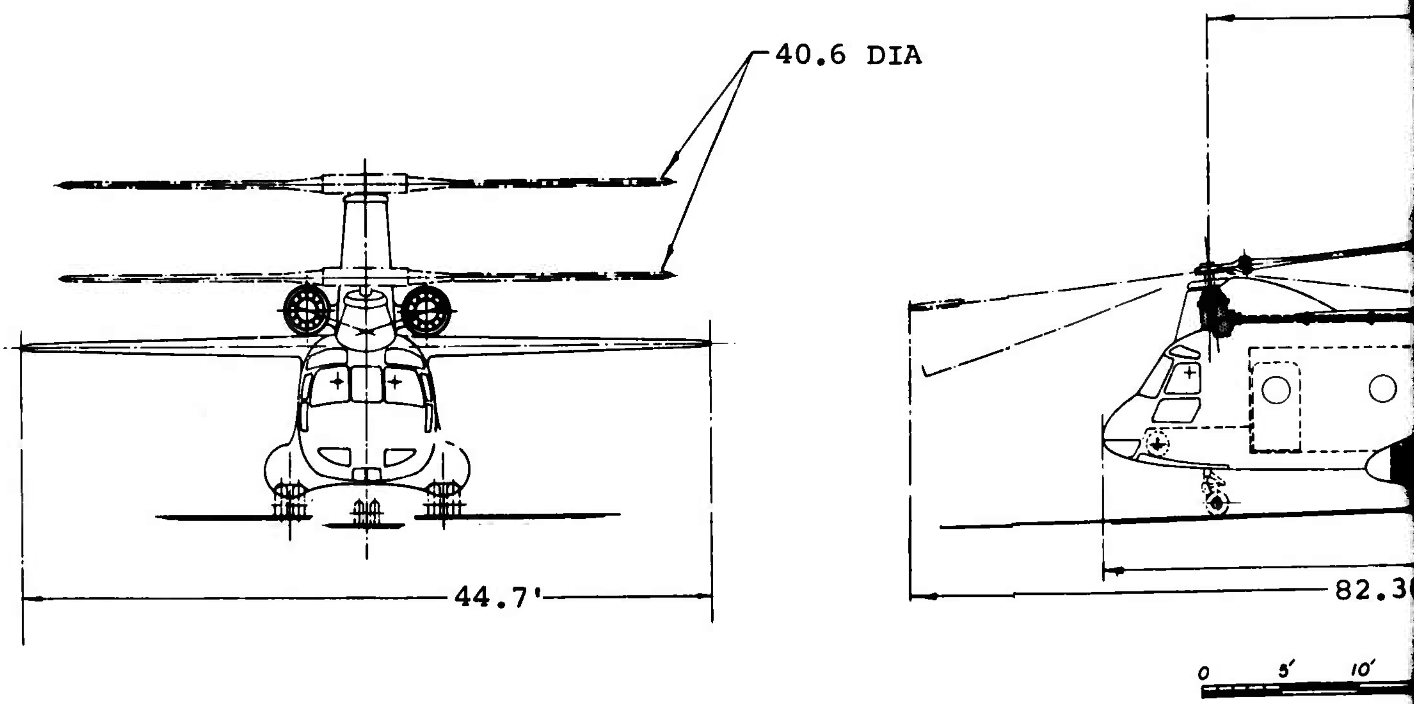


Figure 115. Tandem Rotor Propulsion-Unloaded Compound Aircraft Propulsion System 2a



BASIC DATA

1. TWO ENGINES - EACH 3130 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 4420-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 333 SQ FT
5. DESIGN GROSS WEIGHT - 28,500 LB (LF=3.0)
6. EMPTY WEIGHT - 18,700 LB



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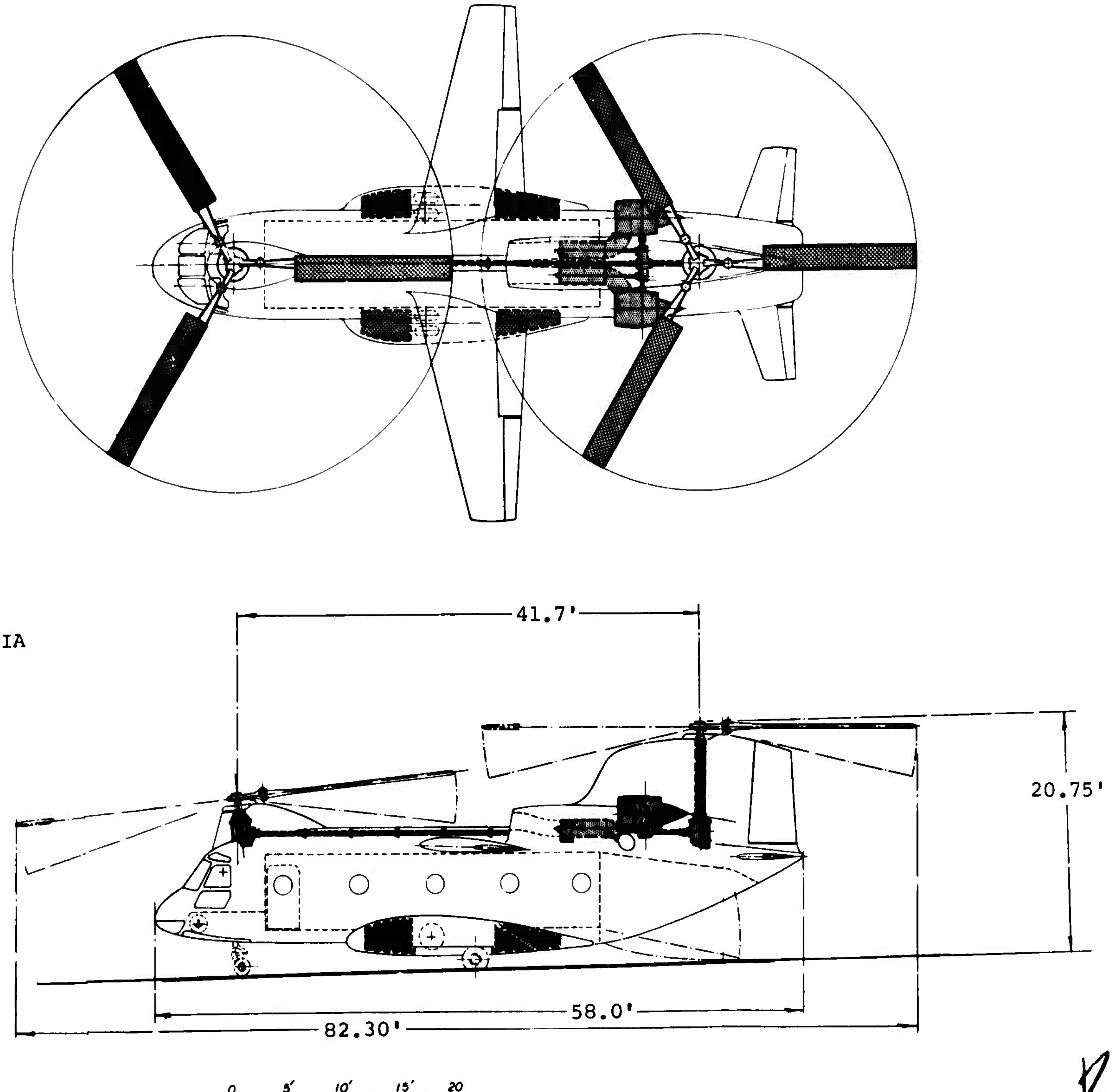

B

Figure 116. Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2a

BASIC DATA

1. TWO ENGINES - EACH 4135 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 5880-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 439 SQ FT
5. DESIGN GROSS WEIGHT - 35,120 LB (LF=3.0)
6. EMPTY WEIGHT - 25,310 LB

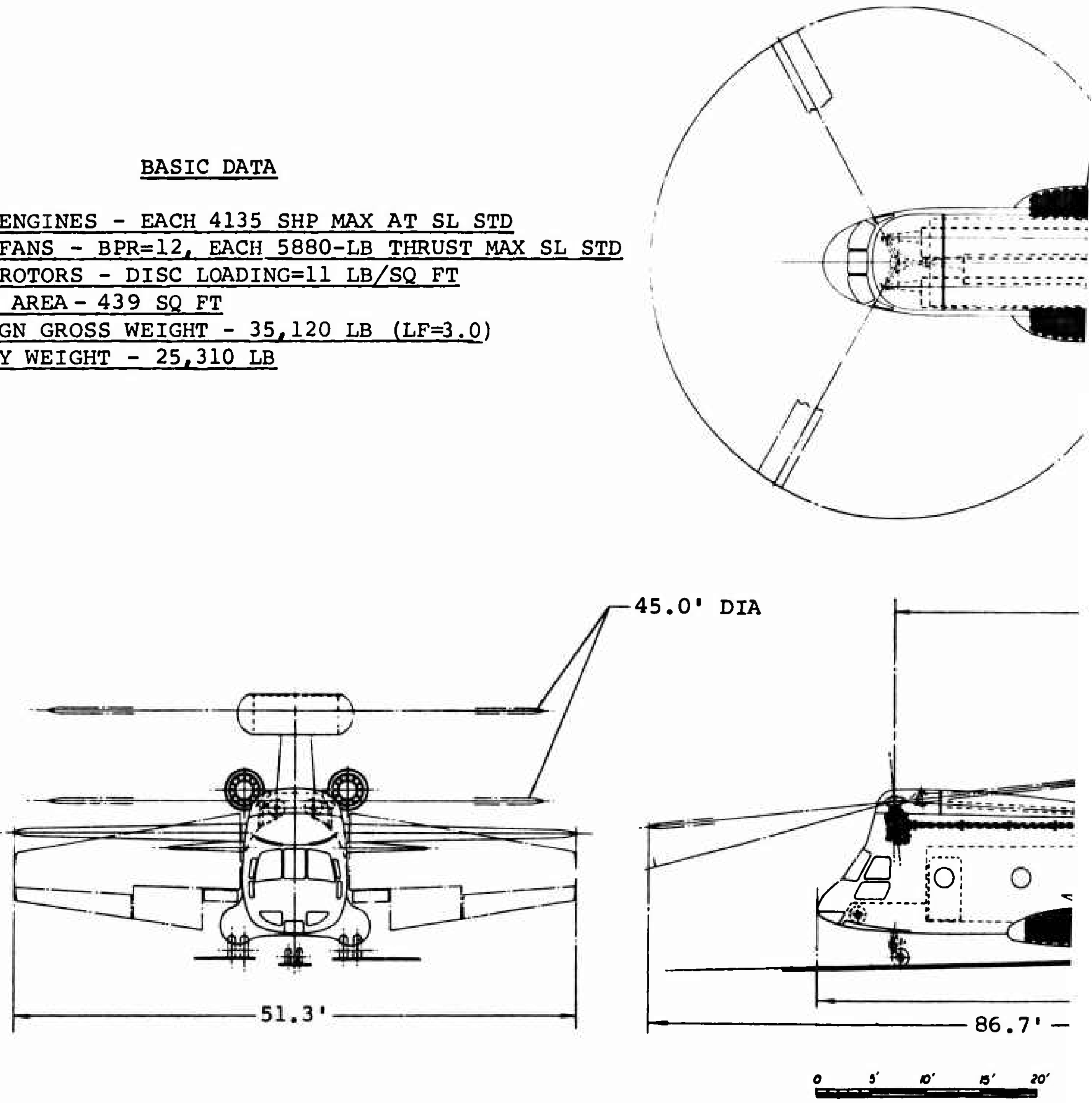

Figure 117. Tandem Rotor Composite Aircraft Propulsion System 2a

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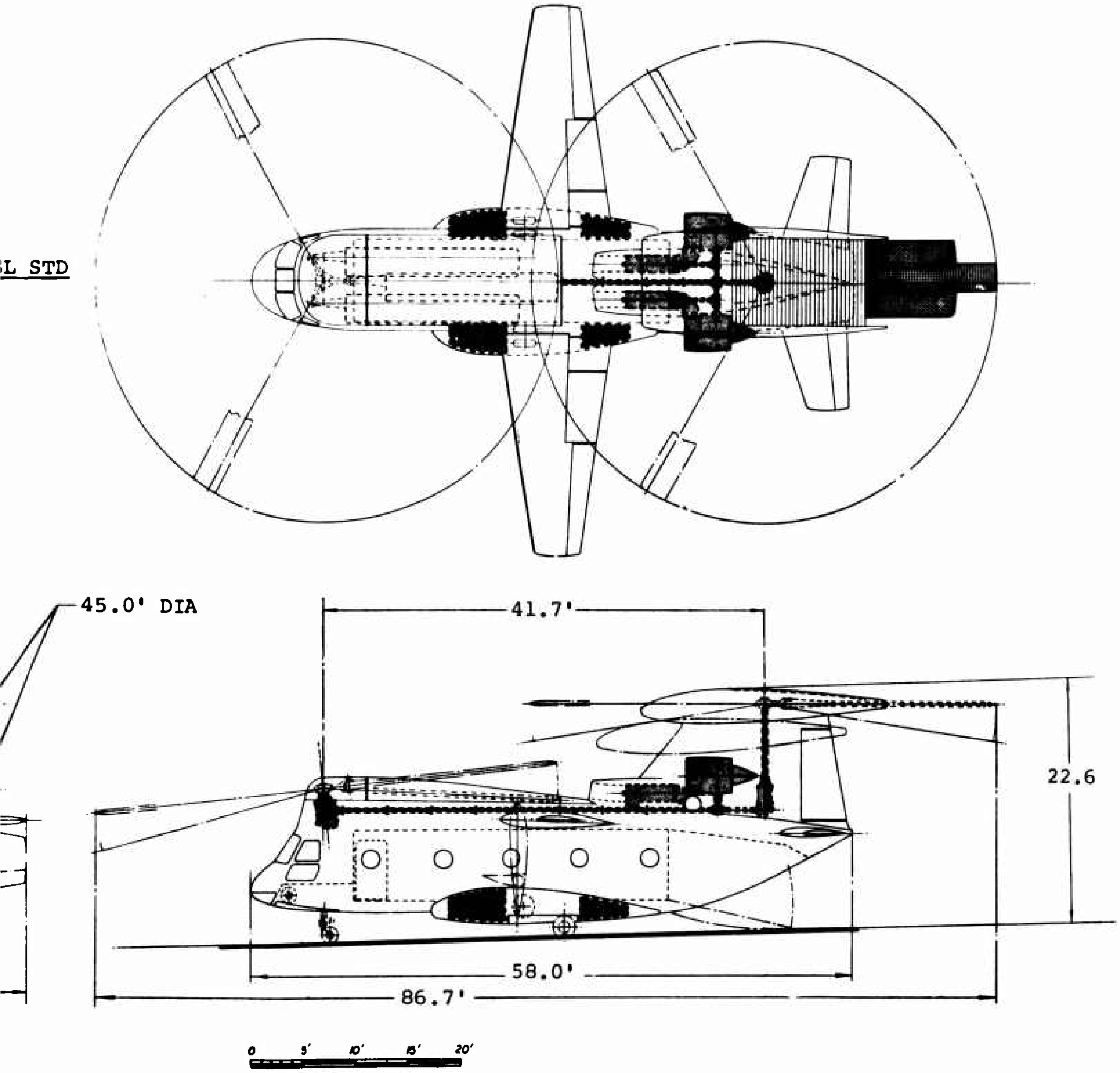

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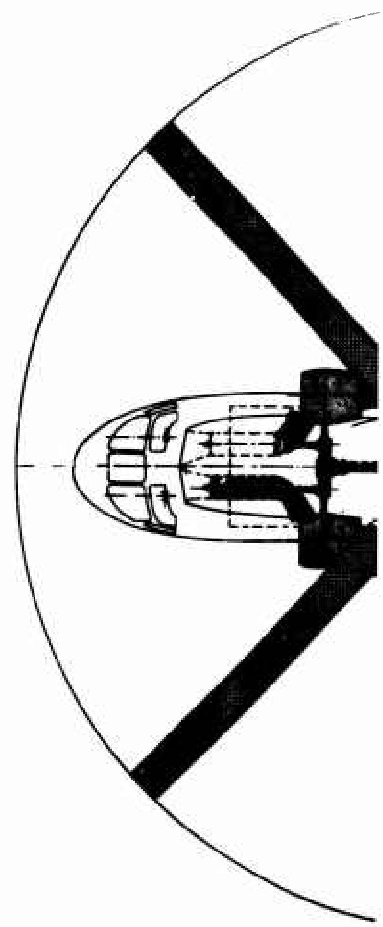
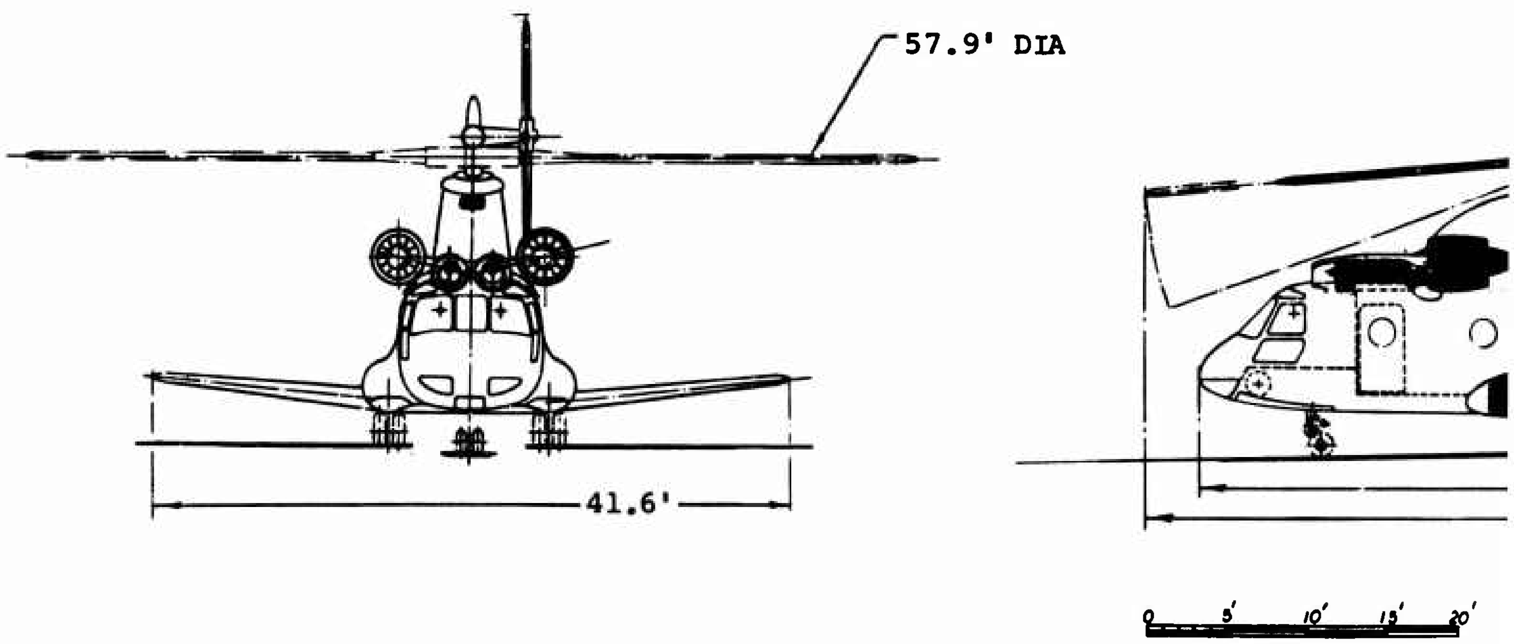


BASIC DATA

1. TWO ENGINES - EACH 3525 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 4980-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 289 SQ FT
5. DESIGN GROSS WEIGHT - 29,030 LB (LF = 3.0)
6. EMPTY WEIGHT - 19,030 LB



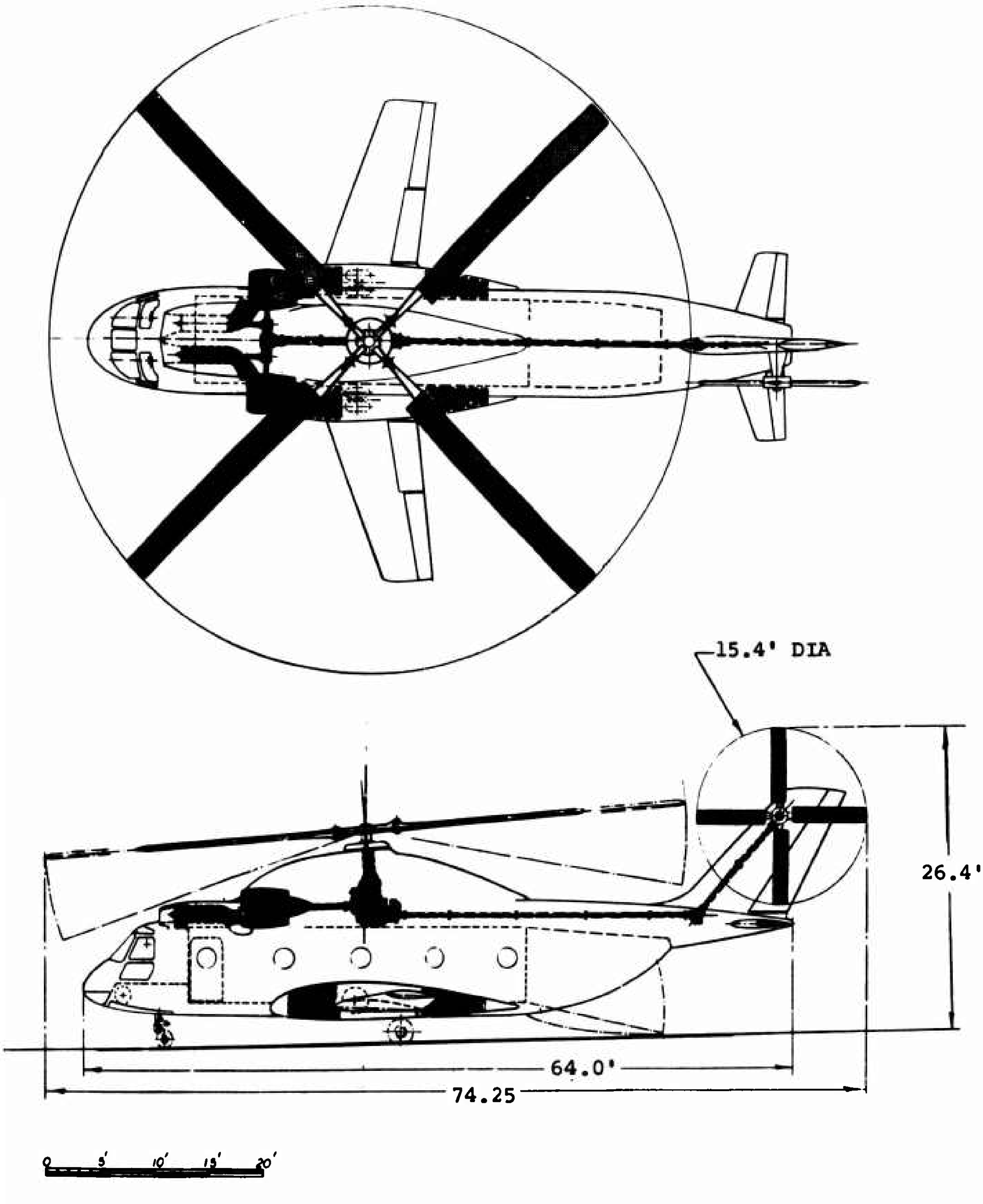
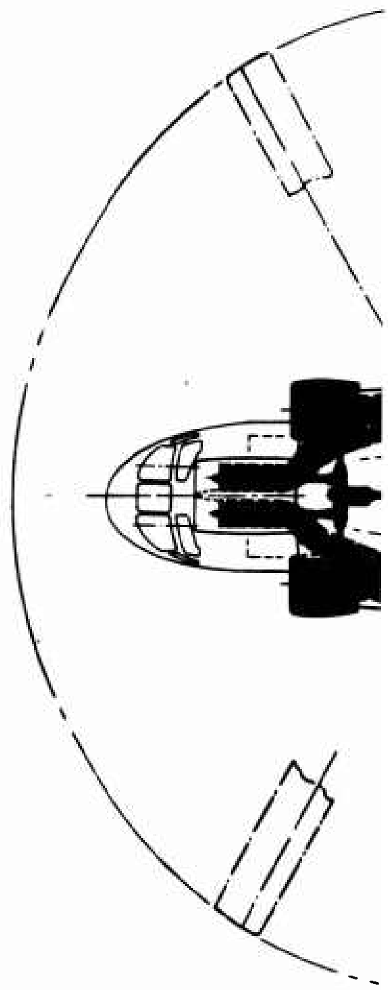
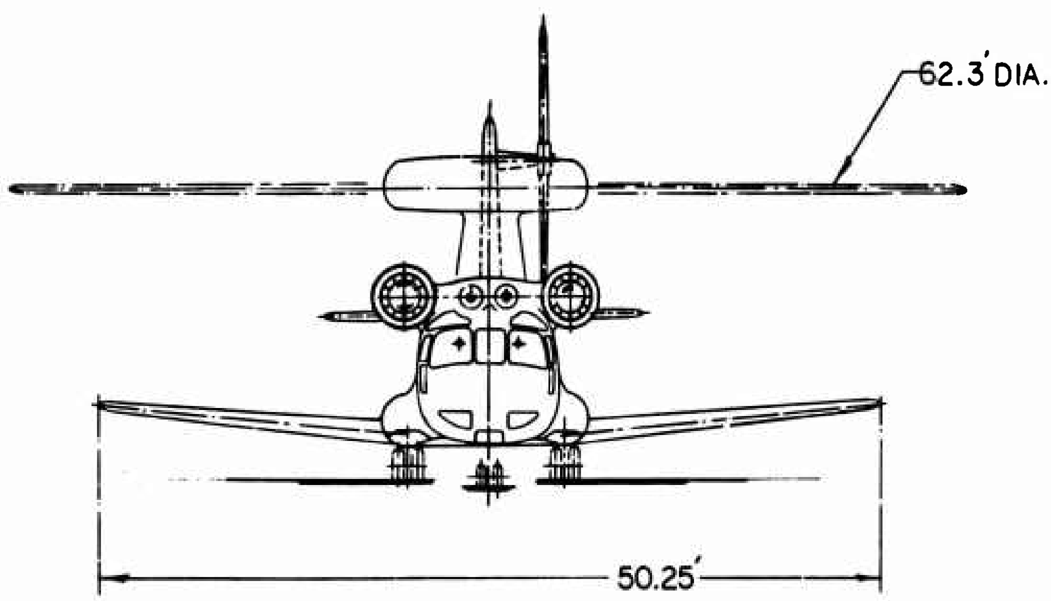


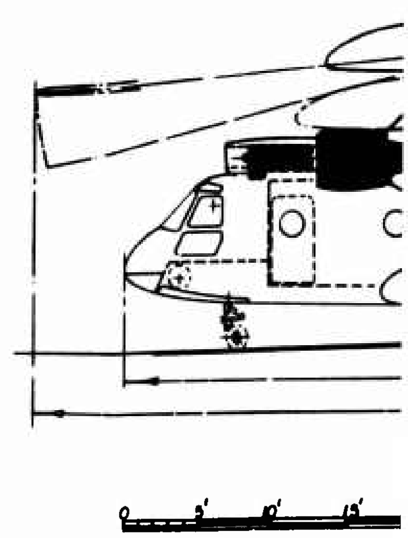
Figure 118. Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2a

BASIC DATA

1. TWO ENGINES - EACH 4105 SHP MAX AT SL STD
2. TWO FANS - BPR=12, EACH 5800-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 421 SQ FT
5. DESIGN GROSS WEIGHT - 33,700 LB (LF=3.0)
6. EMPTY WEIGHT - 23,750 LB







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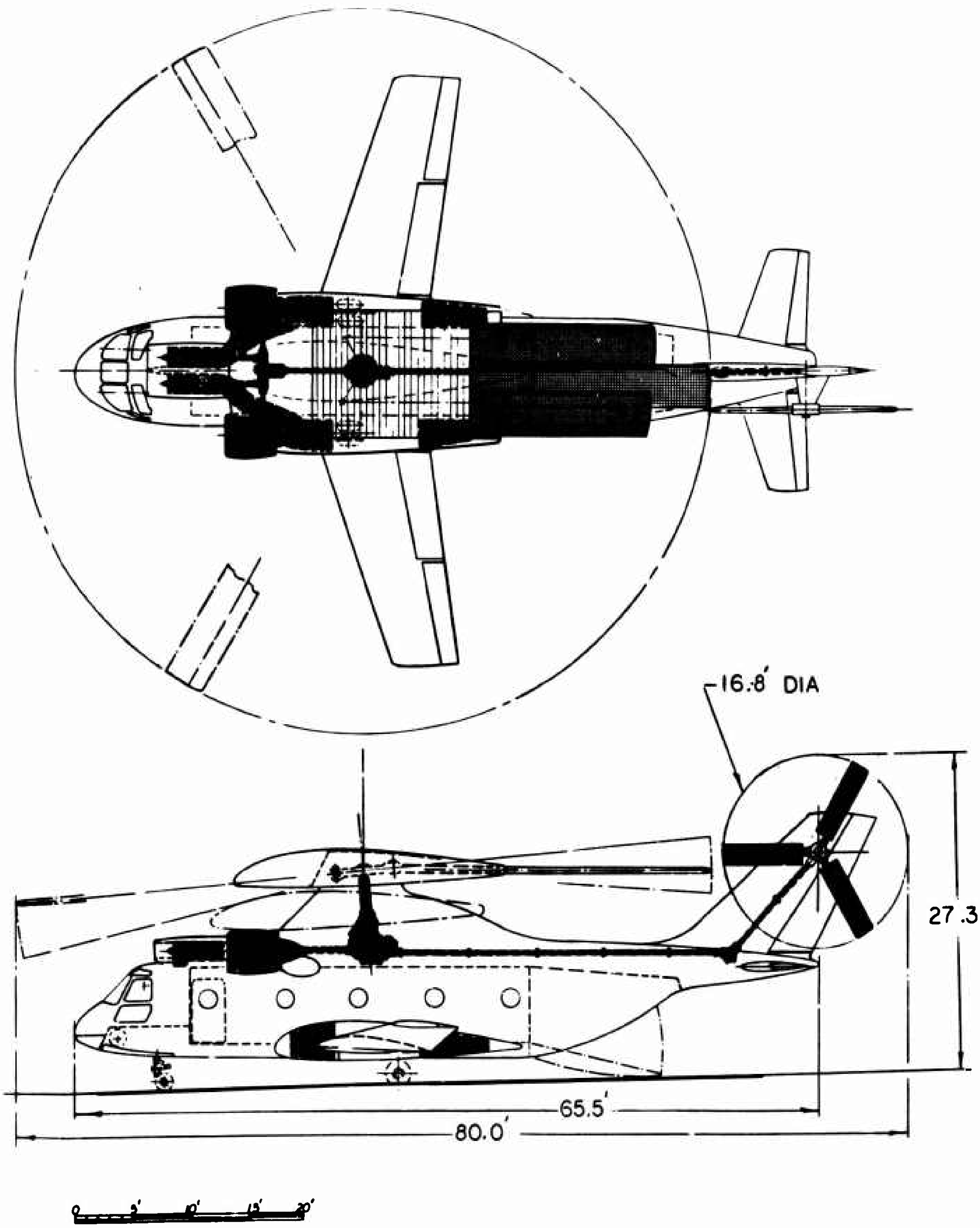
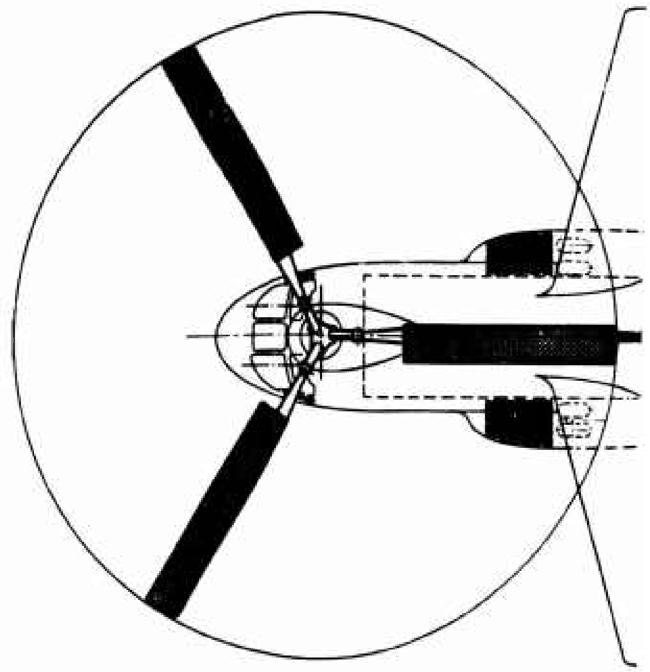


Figure 119. Single Rotor Composite Aircraft Propulsion System 2a



BASIC DATA

1. TWO ENGINES - EACH 3000 SHP MAX AT SL STD
2. TWO FANS - BPR=9, EACH 4590-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 261 SQ FT
5. DESIGN GROSS WEIGHT - 27,500 LB (LF=3.0)
6. EMPTY WEIGHT - 17,850 LB

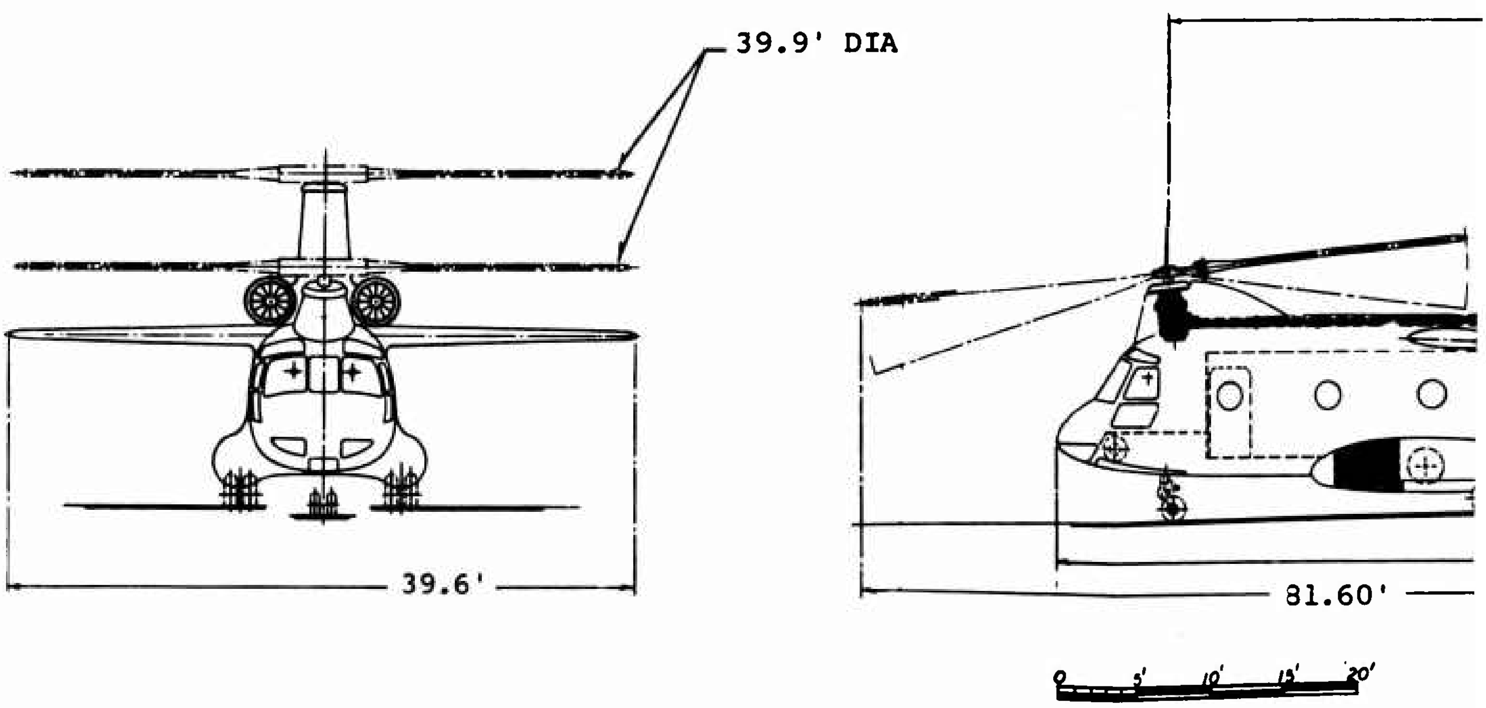
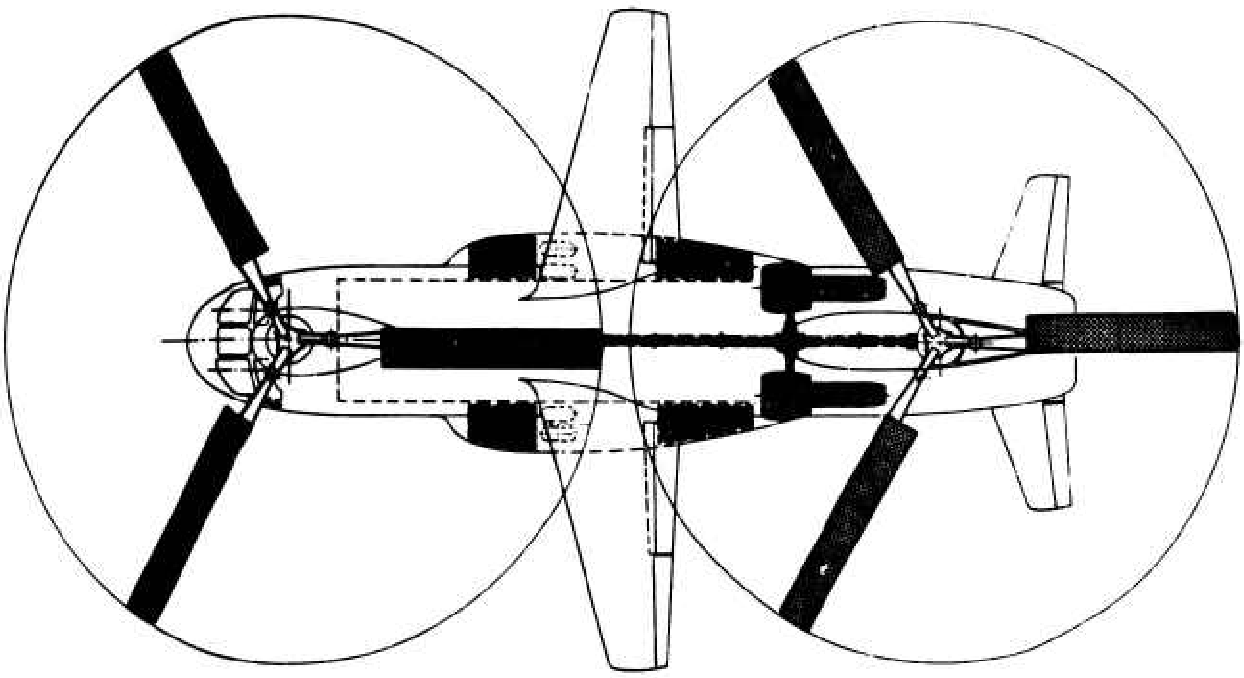


Figure 120. Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2b



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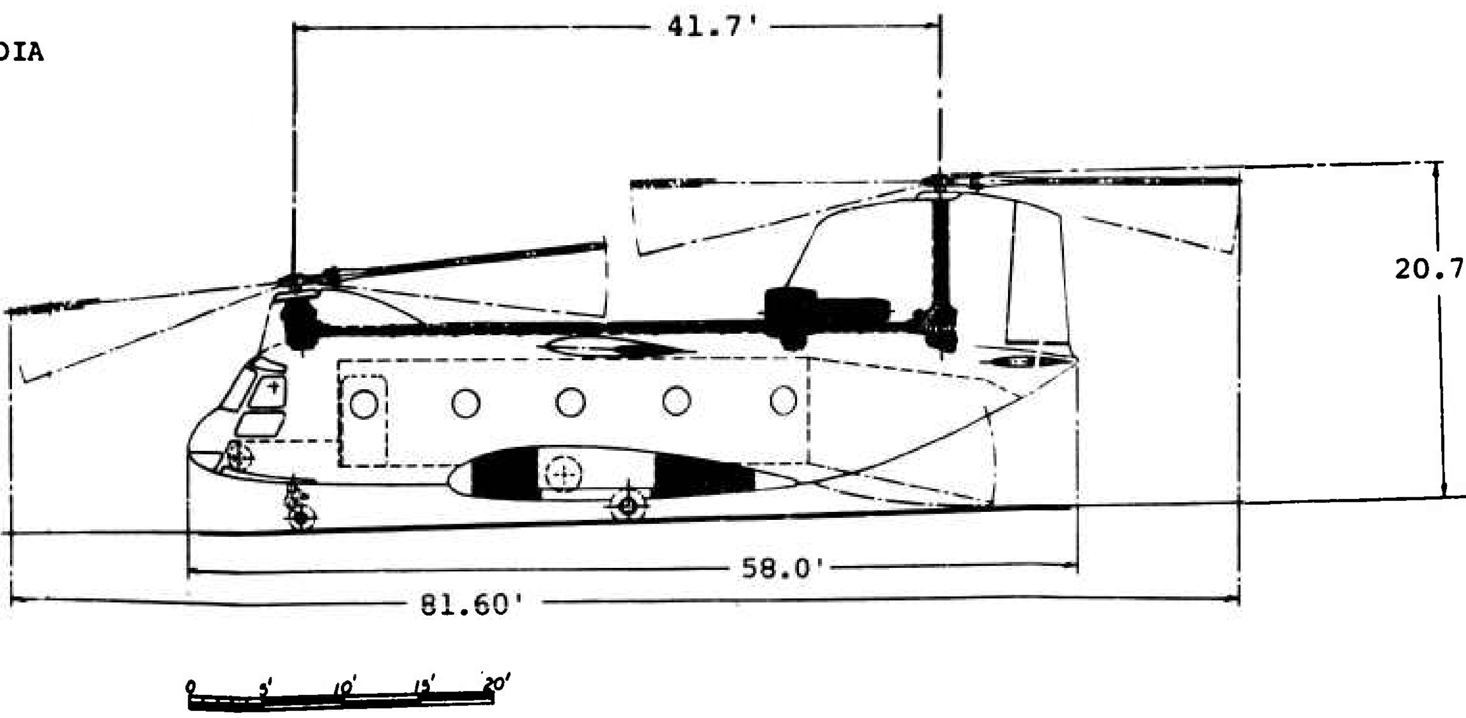

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2b

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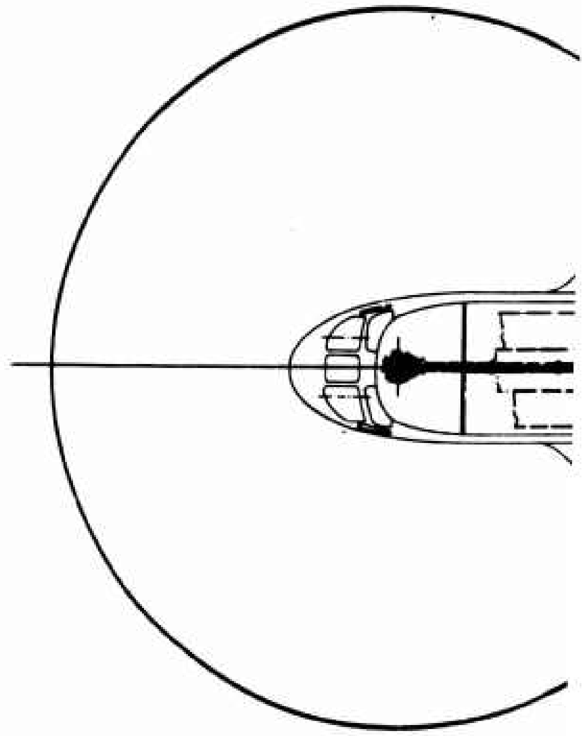
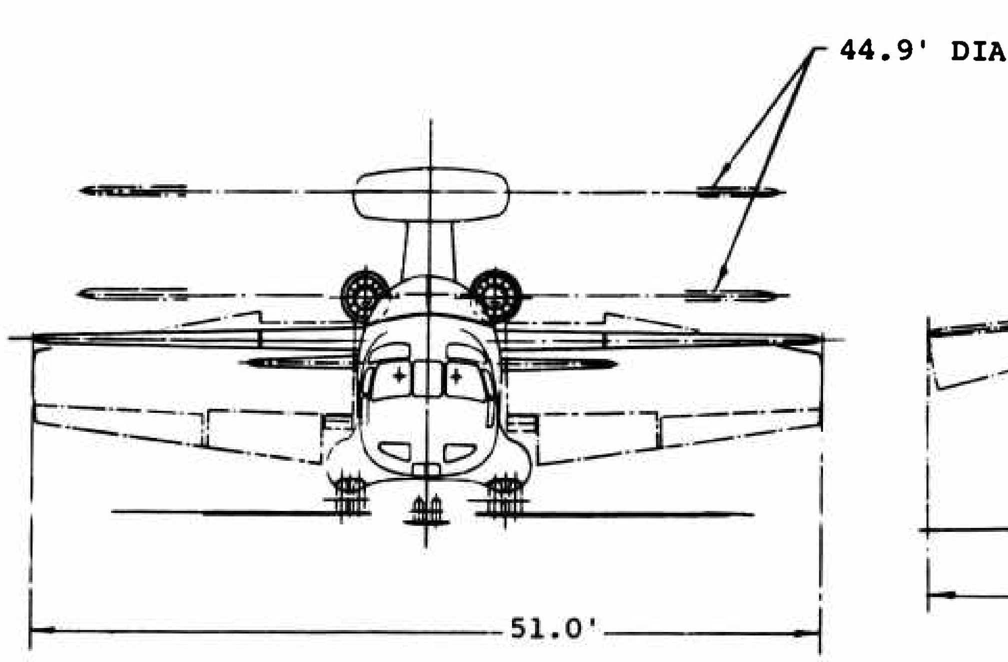
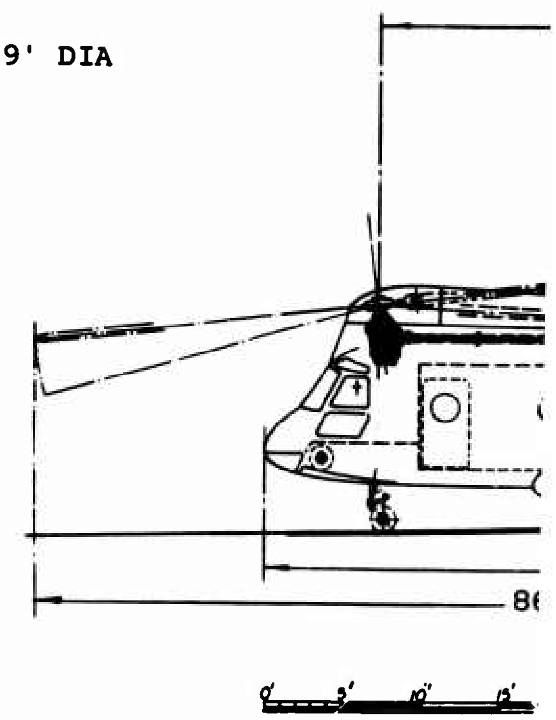


BASIC DATA

1. TWO ENGINES - EACH 4090 SHP MAX AT SL STD
2. TWO FANS - BPR=6, EACH 5600-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=11 LB/SQ FT
4. WING AREA - 434 SQ FT
5. DESIGN GROSS WEIGHT - 34,710 LB (LF=3.0)
6. EMPTY WEIGHT - 24,623 LB





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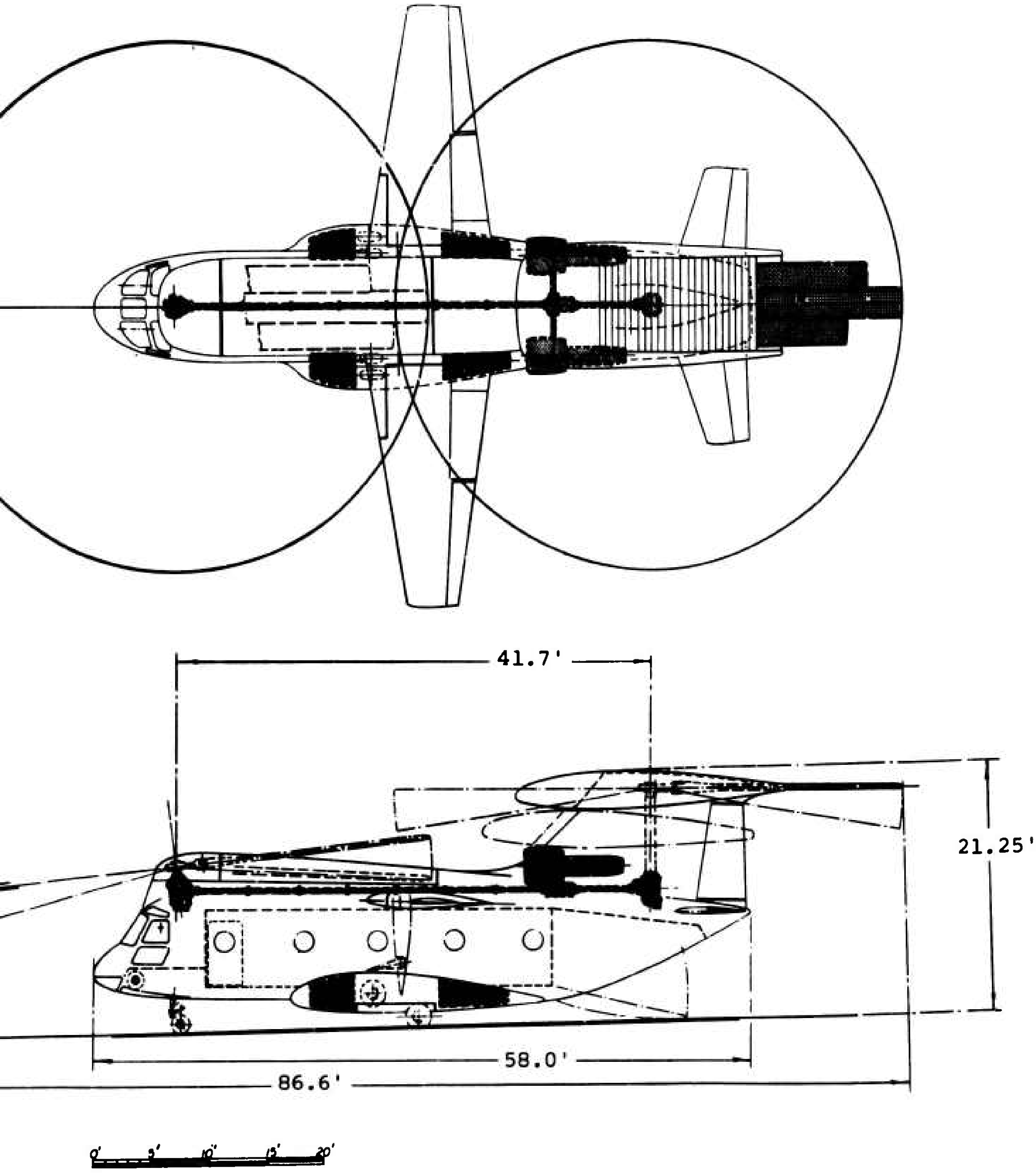
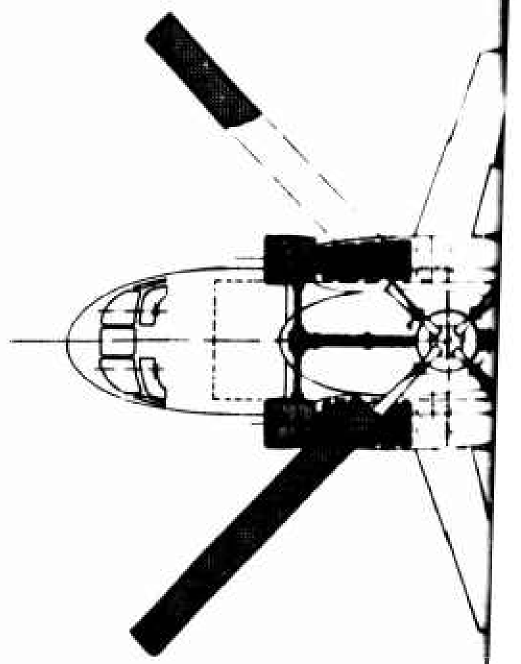


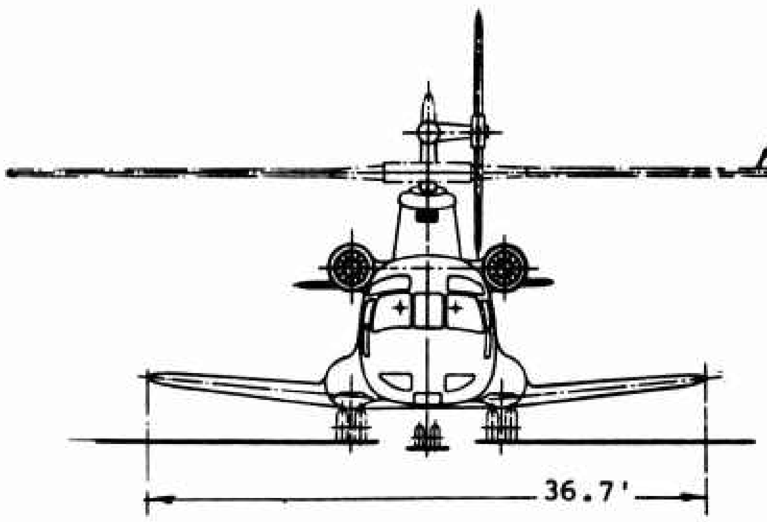
Figure 121. Tandem Rotor Composite Aircraft Propulsion System 2b

B



BASIC DATA

1. TWO ENGINES - EACH 3430 SHP MAX AT SL STD
2. TWO FANS - BPR=9, EACH 5265-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 225 SQ FT
5. DESIGN GROSS WEIGHT - 28,420 LB (LF=3.0)
6. EMPTY WEIGHT - 18,410 LB



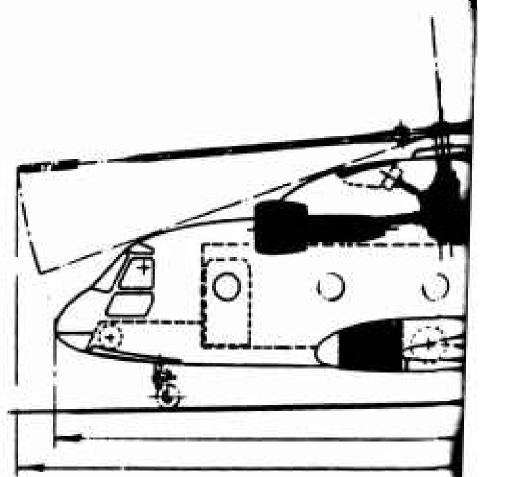


Figure 122. Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 2b

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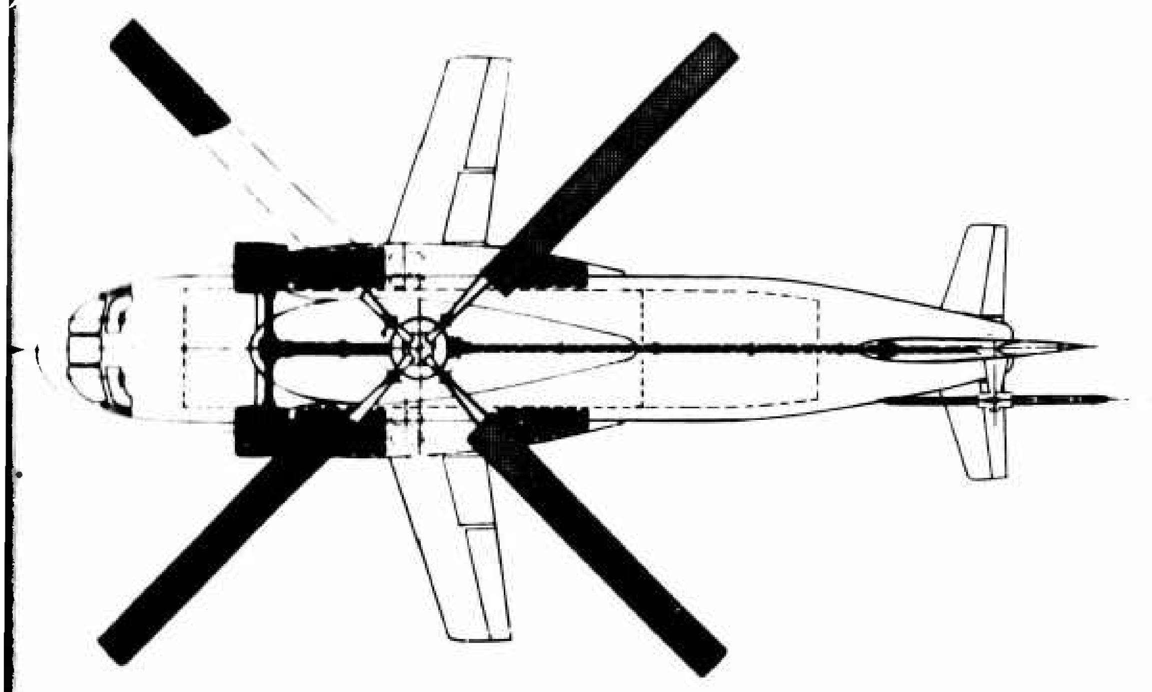
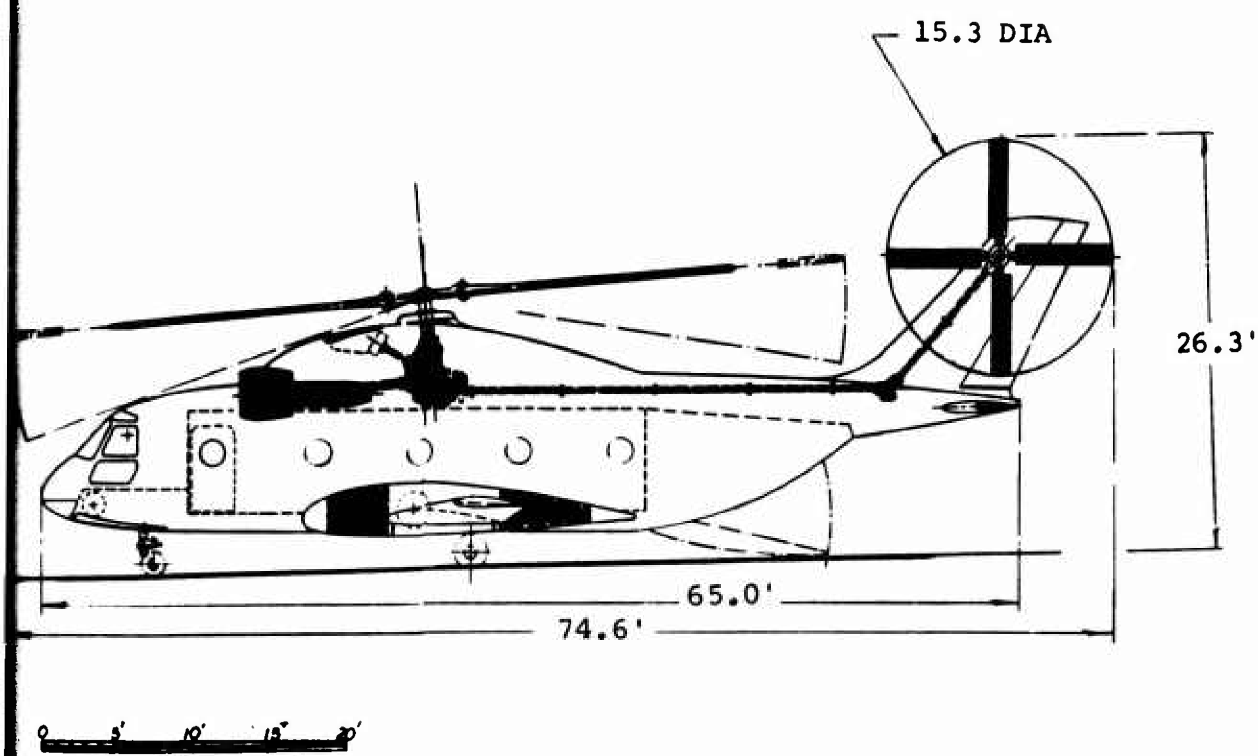
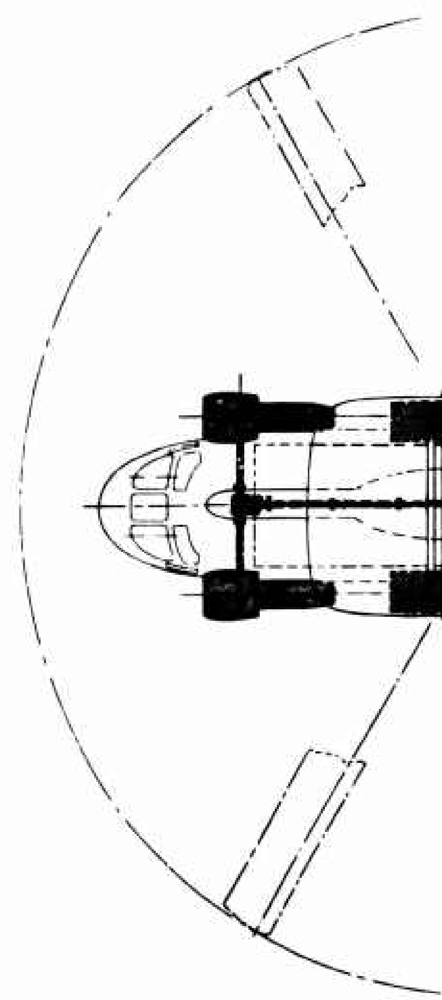




B

BASIC DATA

1. TWO ENGINES - EACH 4090 SHP MAX AT SL STD
2. TWO FANS - BPR=6, EACH 5600-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=11 LB/SQ FT
4. WING AREA - 420 SQ FT
5. DESIGN GROSS WEIGHT - 33,650 LB (LF=3.0)
6. EMPTY WEIGHT - 23,580 LB



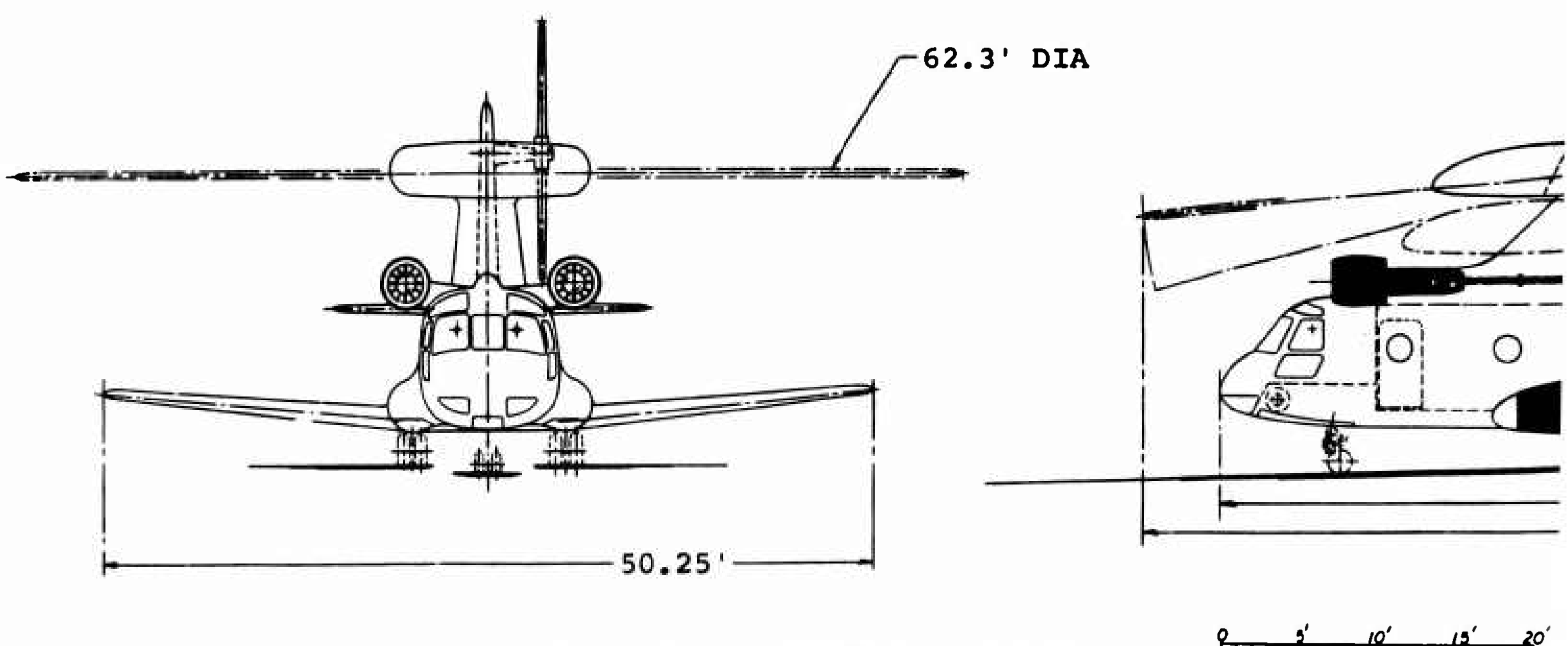


Figure 123. Single Rotor Composite Aircraft Propulsion System 2b

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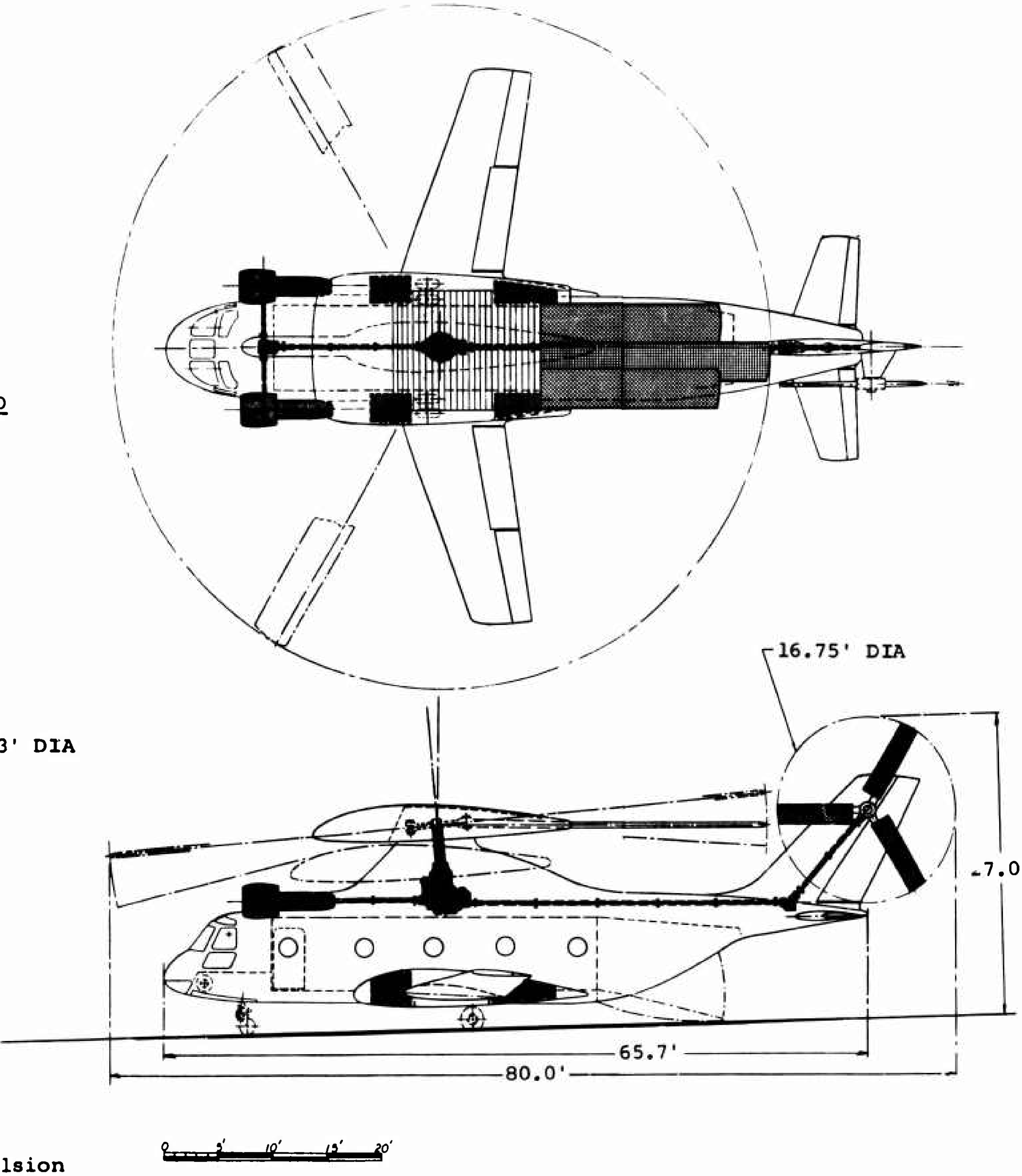


: Propulsion

B

BASIC DATA

1. TWO ENGINES - EACH 3200 SHP MAX AT SL STD
2. TWO FANS - BPR=3, EACH 5020-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=8 LB/SQ FT
4. WING AREA - 200 SQ FT
5. DESIGN GROSS WEIGHT - 31,280 LB (LF=3.0)
6. EMPTY WEIGHT - 20,395 LB

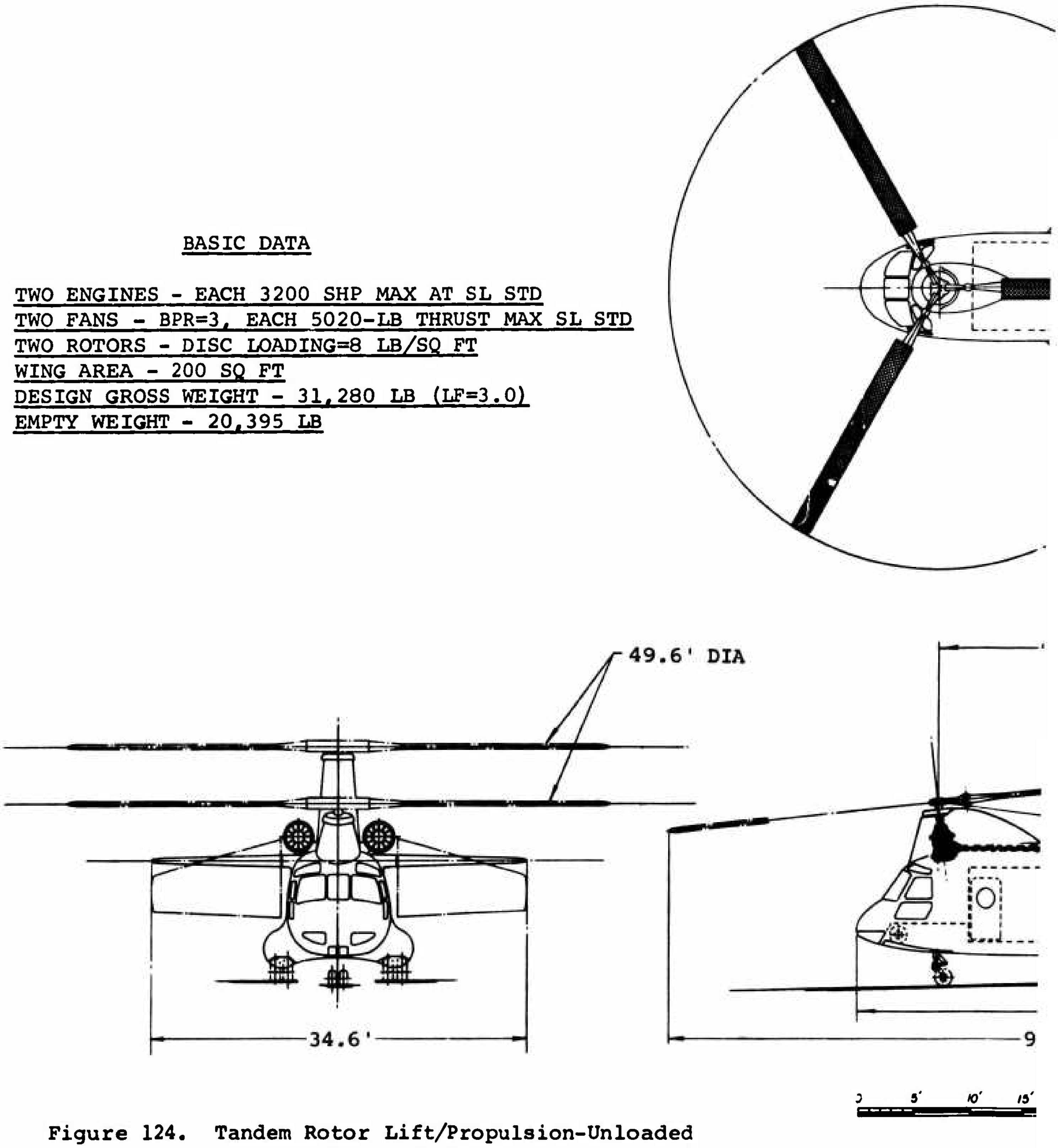
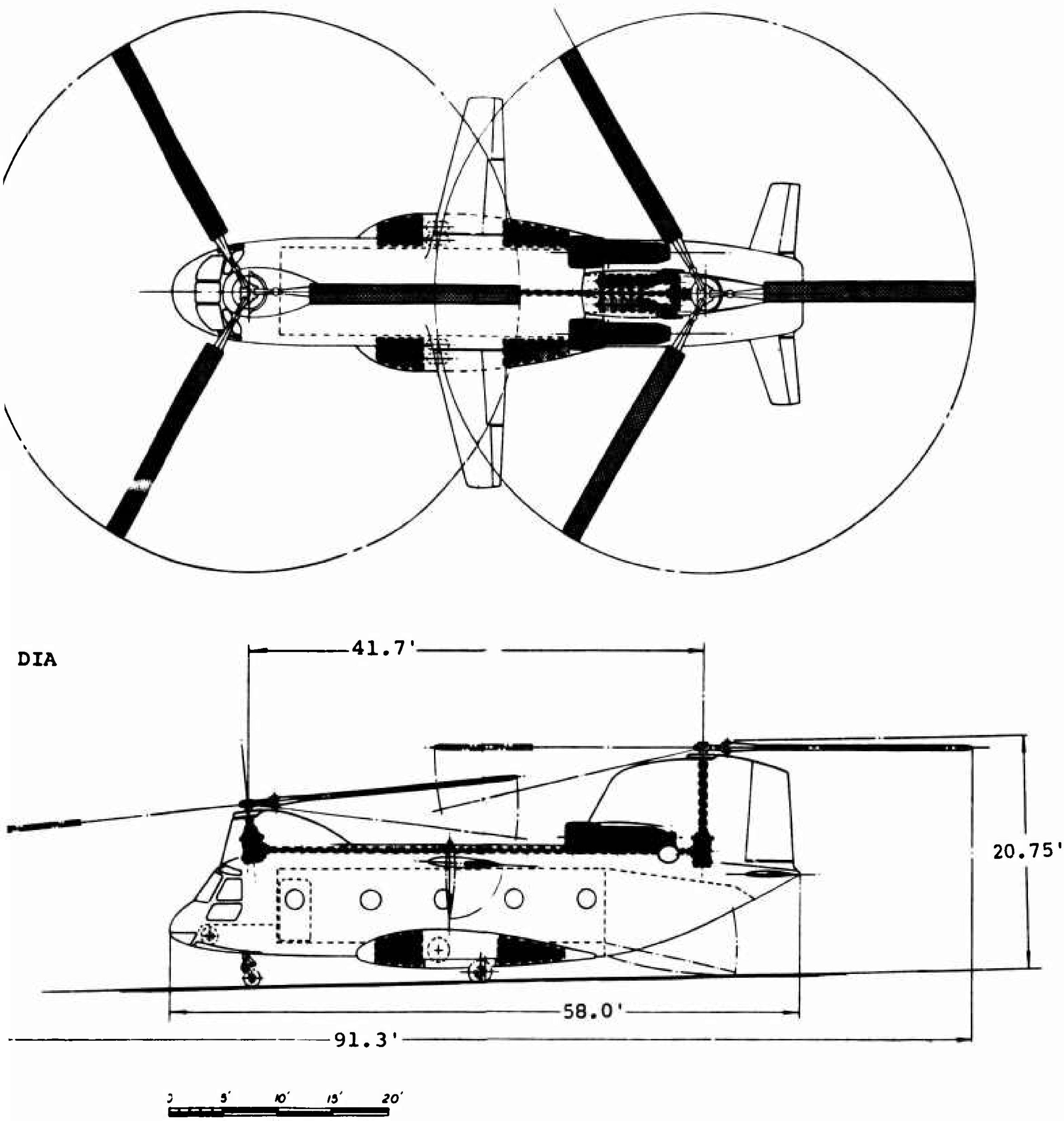


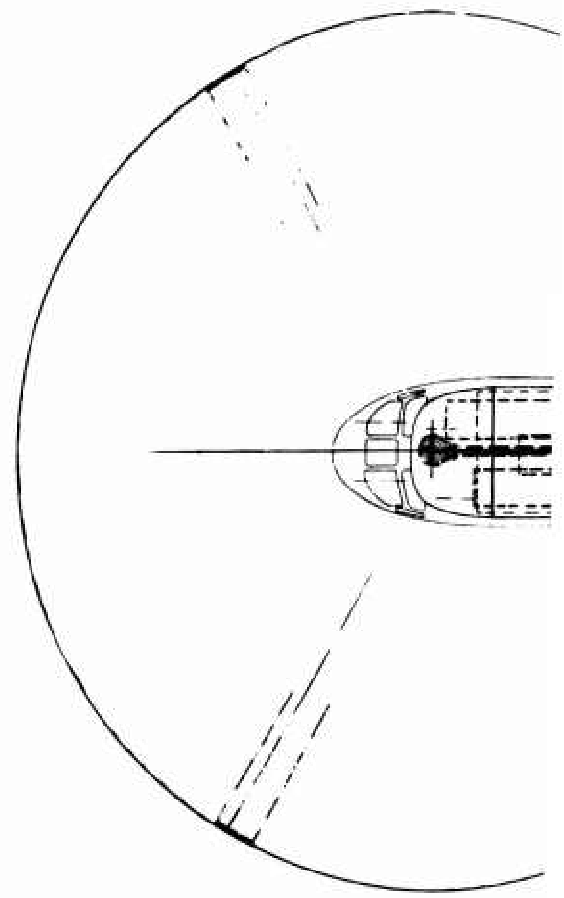
Figure 124. Tandem Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 3

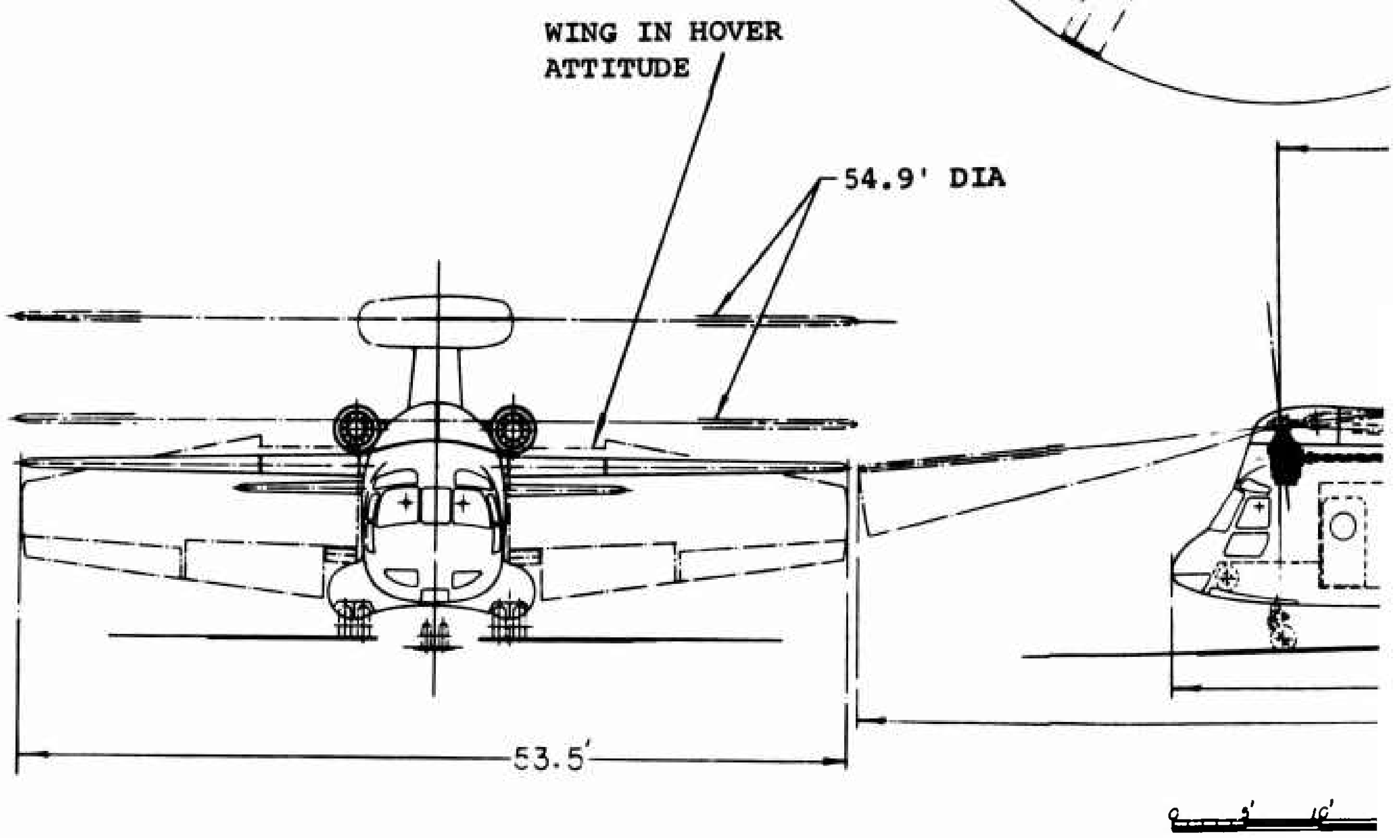


B

BASIC DATA

1. TWO ENGINES - EACH 3960 SHP MAX AT SL STD
2. TWO FANS - BPR=3, EACH 6140-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=8 LB/SQ FT
4. WING AREA - 476 SQ FT
5. DESIGN GROSS WEIGHT - 38,120 LB (LF=3.0)
6. EMPTY WEIGHT - 27,375 LB





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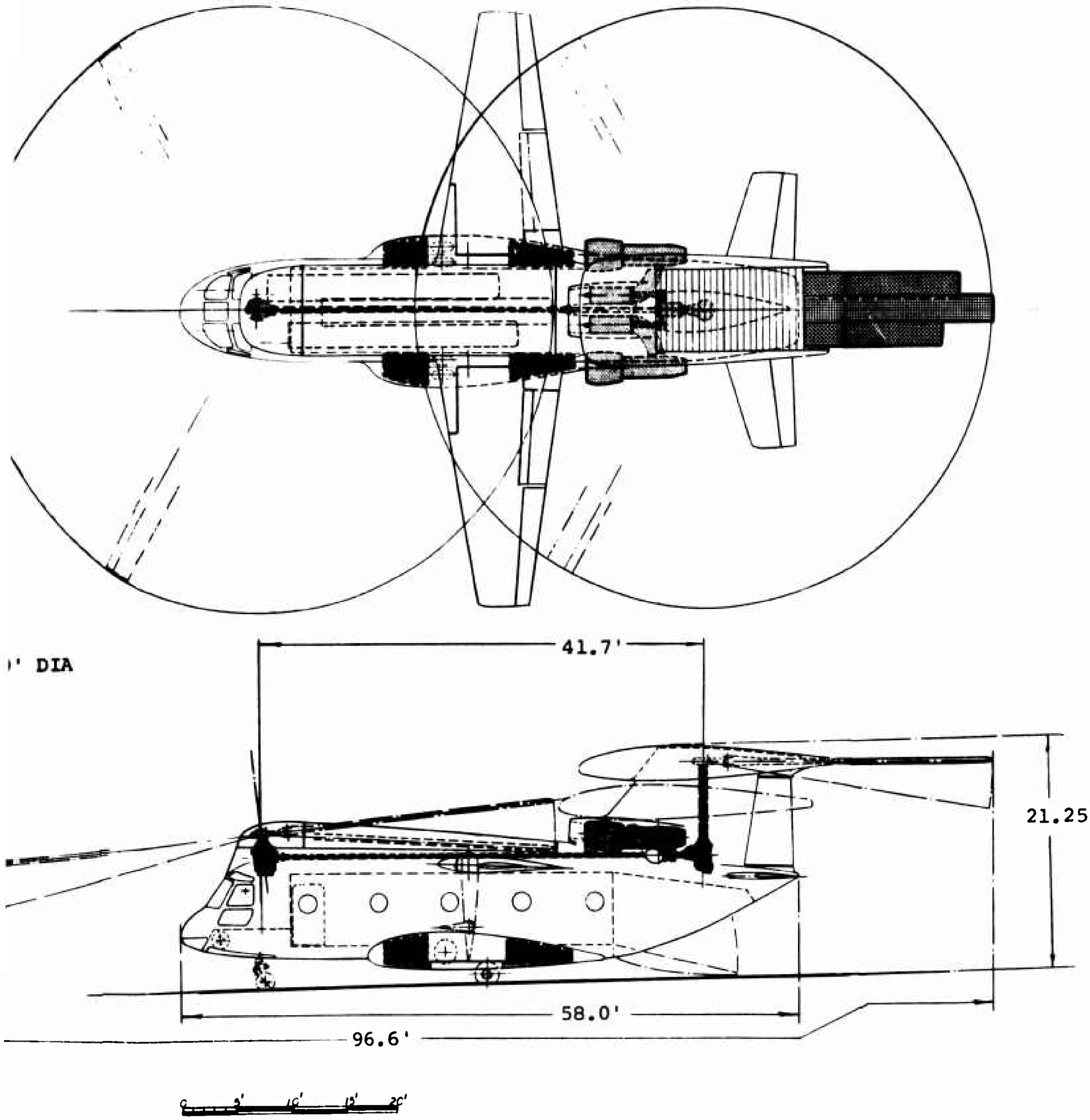
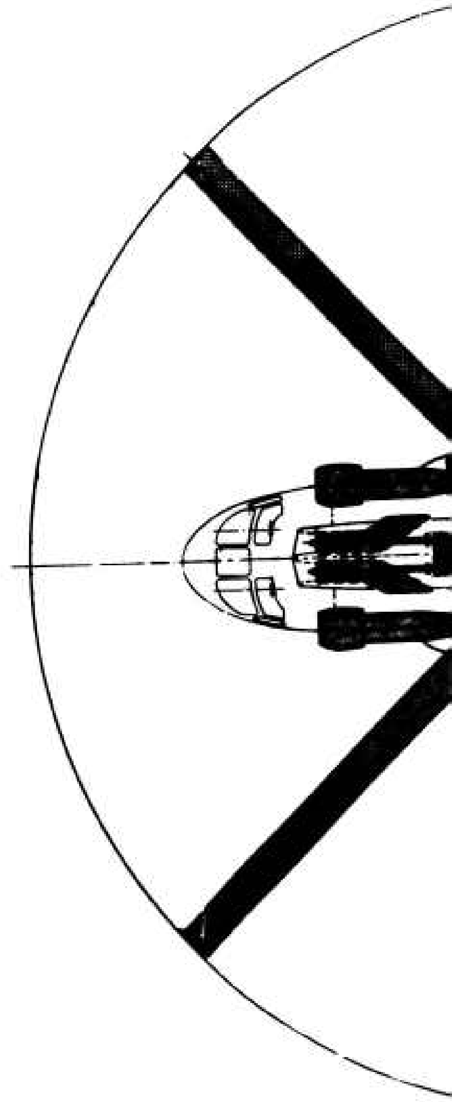
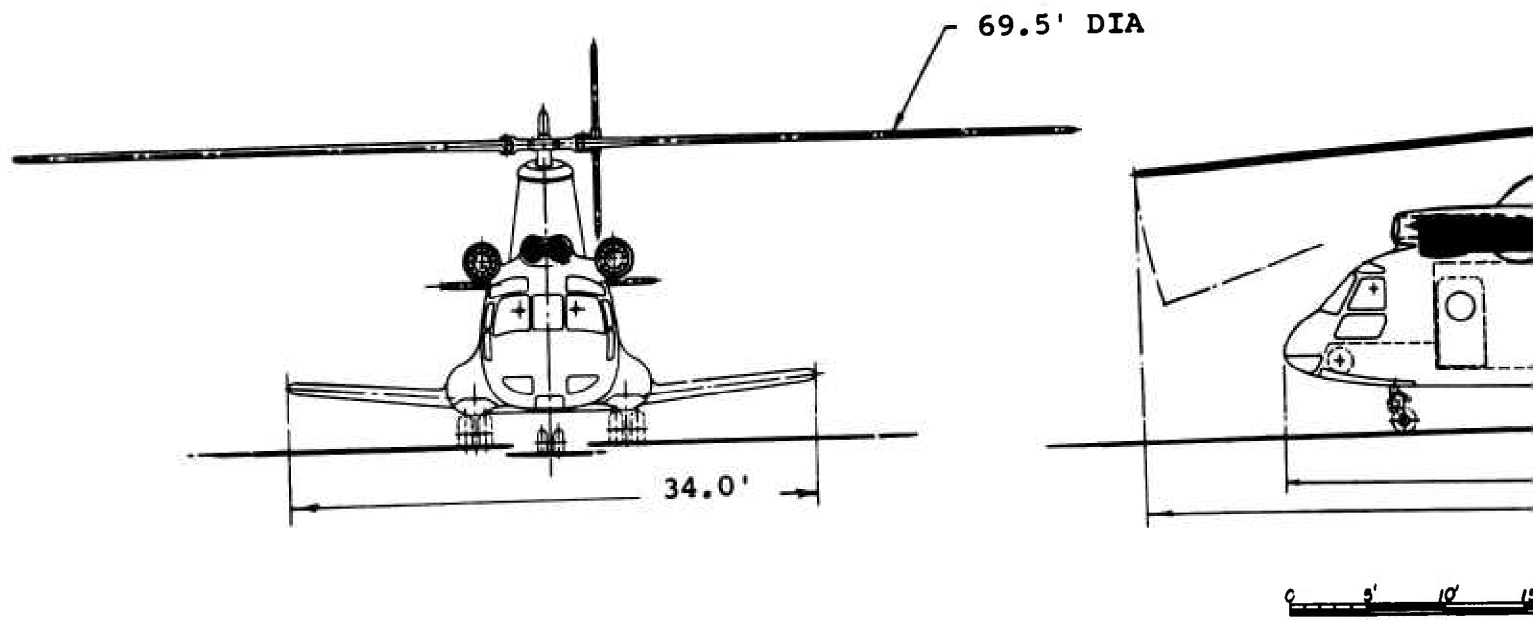


Figure 125. Tandem Rotor Composite Aircraft Propulsion System 3

BASIC DATA

1. TWO ENGINES - EACH 3090 SHP MAX AT SL STD
2. TWO FANS - BPR=3, EACH 4940-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=8 LB/SQ FT
4. WING AREA - 194 SQ FT
5. DESIGN GROSS WEIGHT - 30,520 LB (LF=3.0)
6. EMPTY WEIGHT - 19,732 LB





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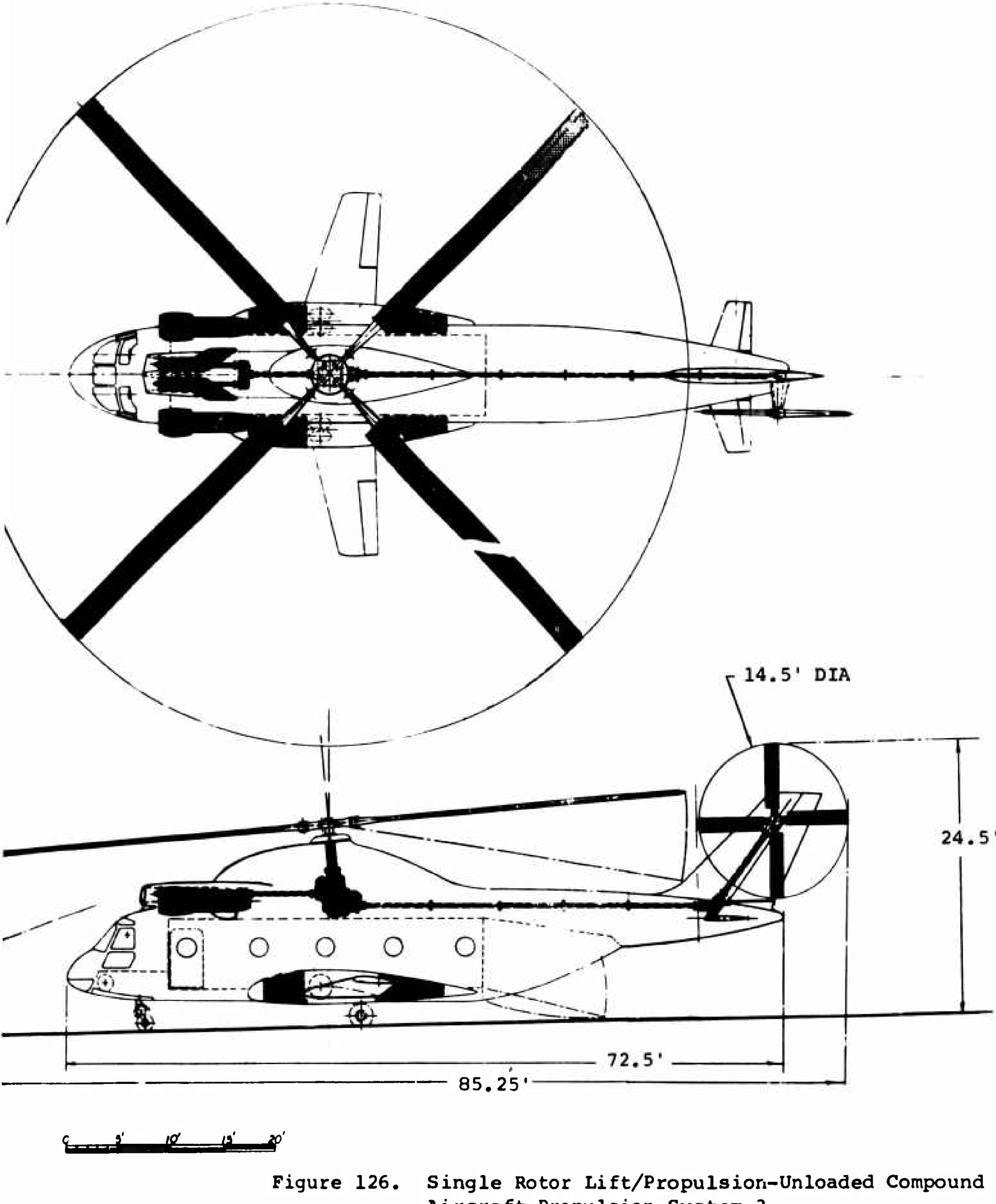
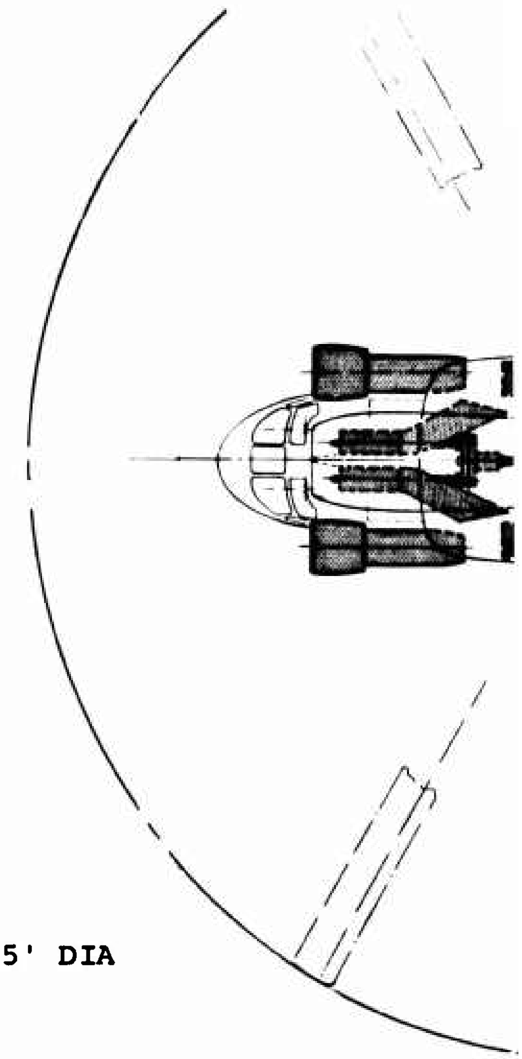


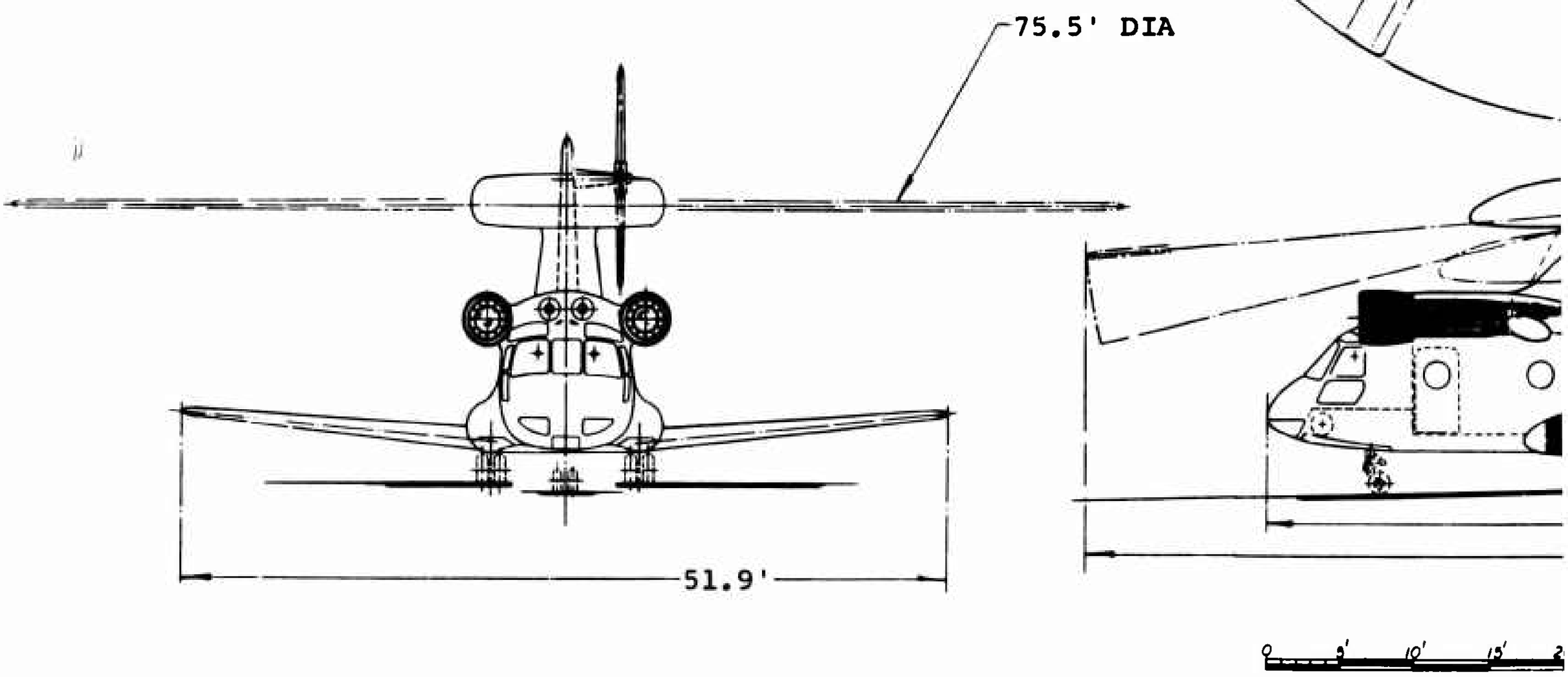
Figure 126. Single Rotor Lift/Propulsion-Unloaded Compound Aircraft Propulsion System 3

B

BASIC DATA

1. TWO ENGINES - EACH 3685 SHP MAX AT SL STD
2. TWO FANS - BPR=3, EACH 6100-LB THRUST MAX SL STD
3. ONE MAIN ROTOR - DISC LOADING=8 LB/SQ FT
4. WING AREA - 449 SQ FT
5. DESIGN GROSS WEIGHT - 35,850 LB (LF=3.0)
6. EMPTY WEIGHT - 25,168 LB





A

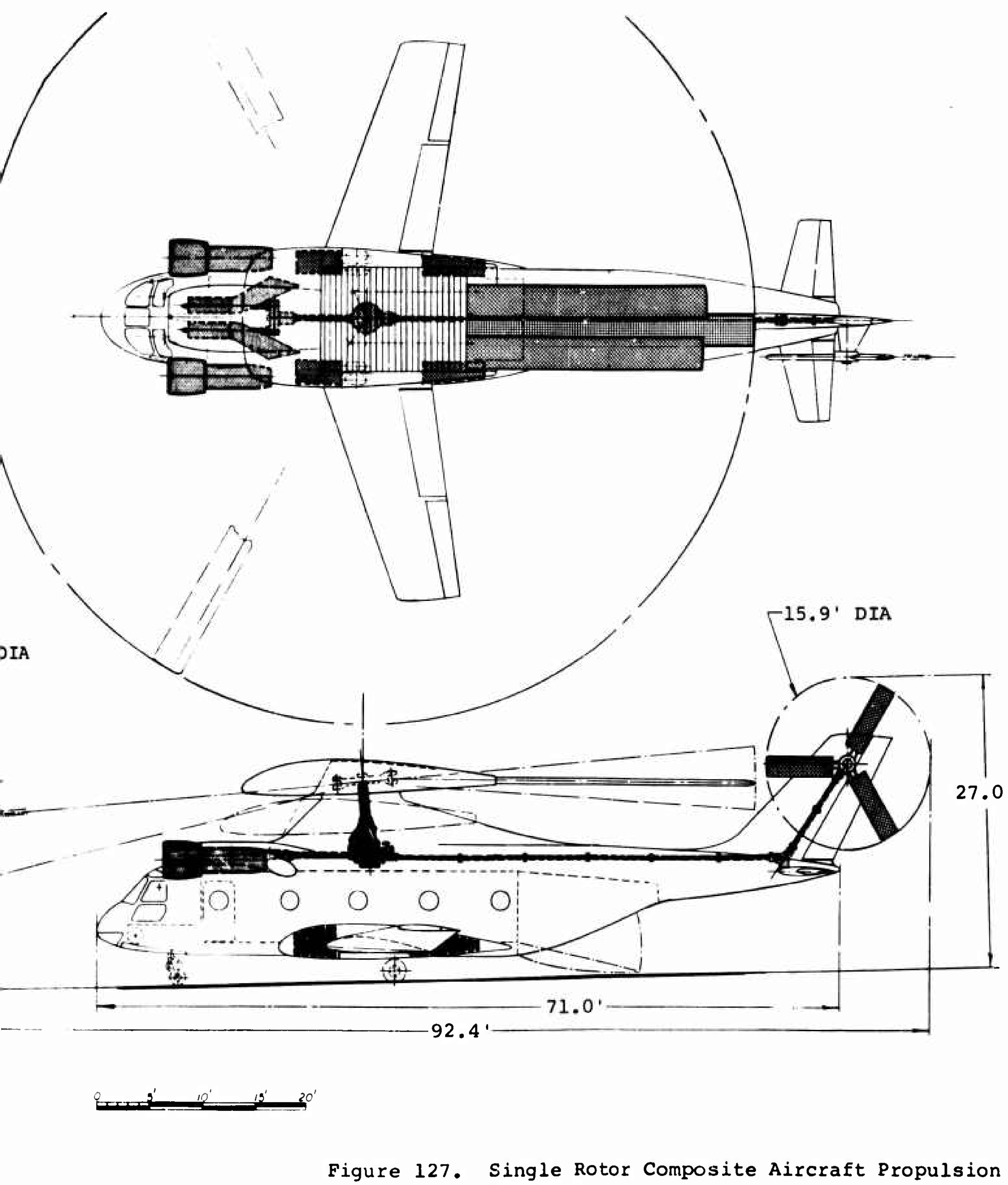


Figure 127. Single Rotor Composite Aircraft Propulsion System 3

B

Propulsion System Drawings - Figures 128 through 137 are drawings of the five propulsion systems associated with both tandem and single rotor aircraft. Included with the figures are propulsion system weight breakdowns keyed by number to the propulsion system components.

TABLE XVIII
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 1a - TANDEM ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb) PU Compound D/L=11 B=12	LPU Compound D/L=11 B=12	Composite D/L=11 B=12
1	Gas generators (2)	325	360	485
2*	Powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	174	182	198
4	Cruise fan assemblies[2] (2)	360	570	855
5	Diverter valves (2)	60	66	95
6	Gas ducts	33	33	33
7	Remote power turbine assembly[2] (1)	177	200	285
8	Power turbine reduction gearbox (1)	113	122	123
9	Rotor transmission assemblies (2)	1148	1280	1773
10	Synchronizing shaft assembly (1)	184	198	230
11	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	252	281	380
12	Aft rotor shaft	196	234	324
13*	Engine section[3]	(136)	(151)	(184)
14	Rotor systems[3] (2)	(2435)	(2340)	(4500)
	Total propulsion system weight	3155	3659	4914

*Not shown

1 Includes exhaust, starting, controls
2 Includes variable gas admission devices
3 Included here for convenience only; normally part of airframe

Key:
a - Gas-driven tip turbine fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

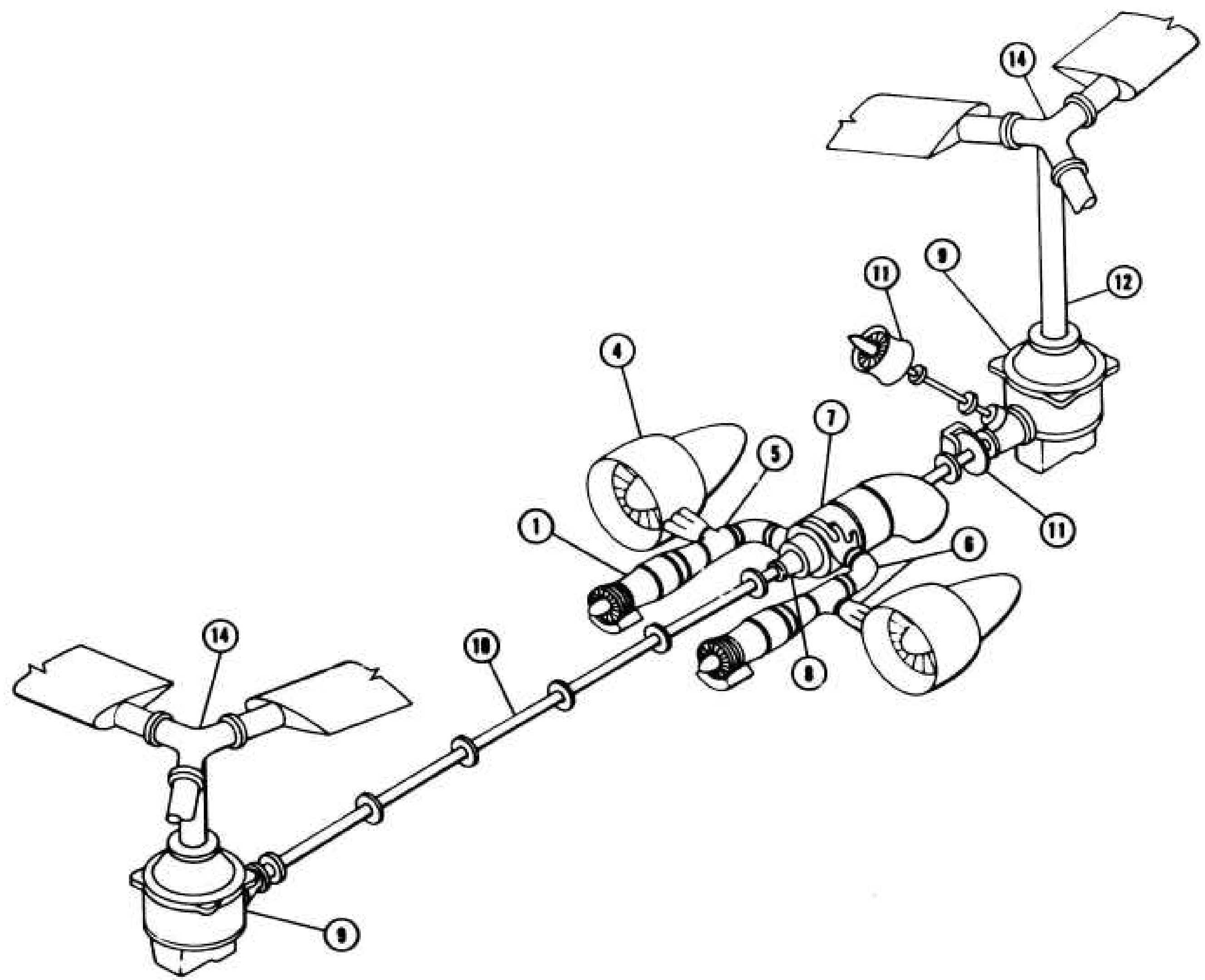


Figure 128. Propulsion System la for Tandem Rotor Aircraft

TABLE XIX
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 1b - TANDEM ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=11 B=9	D/L=11 B=12	D/L=11 B=9
1	Gas generators (2)	325	370	490
2*	Powerplant subgroups (less fuel system)[1] (2)	133	133	133
3*	Fuel system (1)	173	183	194
4	Cruise fan assemblies[2] (2)	230	920	1250
5	Diverter valves (2)	60	71	92
6	Gas ducts	33	33	33
7	Remote power turbine assembly[2] (1)	175	205	285
8	Power turbine reduction gearbox (1)	114	118	122
9	Rotor transmission assemblies (2)	1120	1318	1786
10	Synchronizing shaft assembly (1)	184	198	231
11	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	248	287	382
12	Aft rotor shaft	192	243	325
13*	Engine section[3]	(130)	(164)	(196)
14	Rotor systems[3] (2)	(2460)	(2410)	(1985)
	Total propulsion system weight	2987	4079	5323

*Not shown
1 Includes induction, exhaust, starting
2 Includes variable gas admission devices
3 Included here for convenience only; normally part of airframe

Key:
1(b) - Gas-driven hub-turbine fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

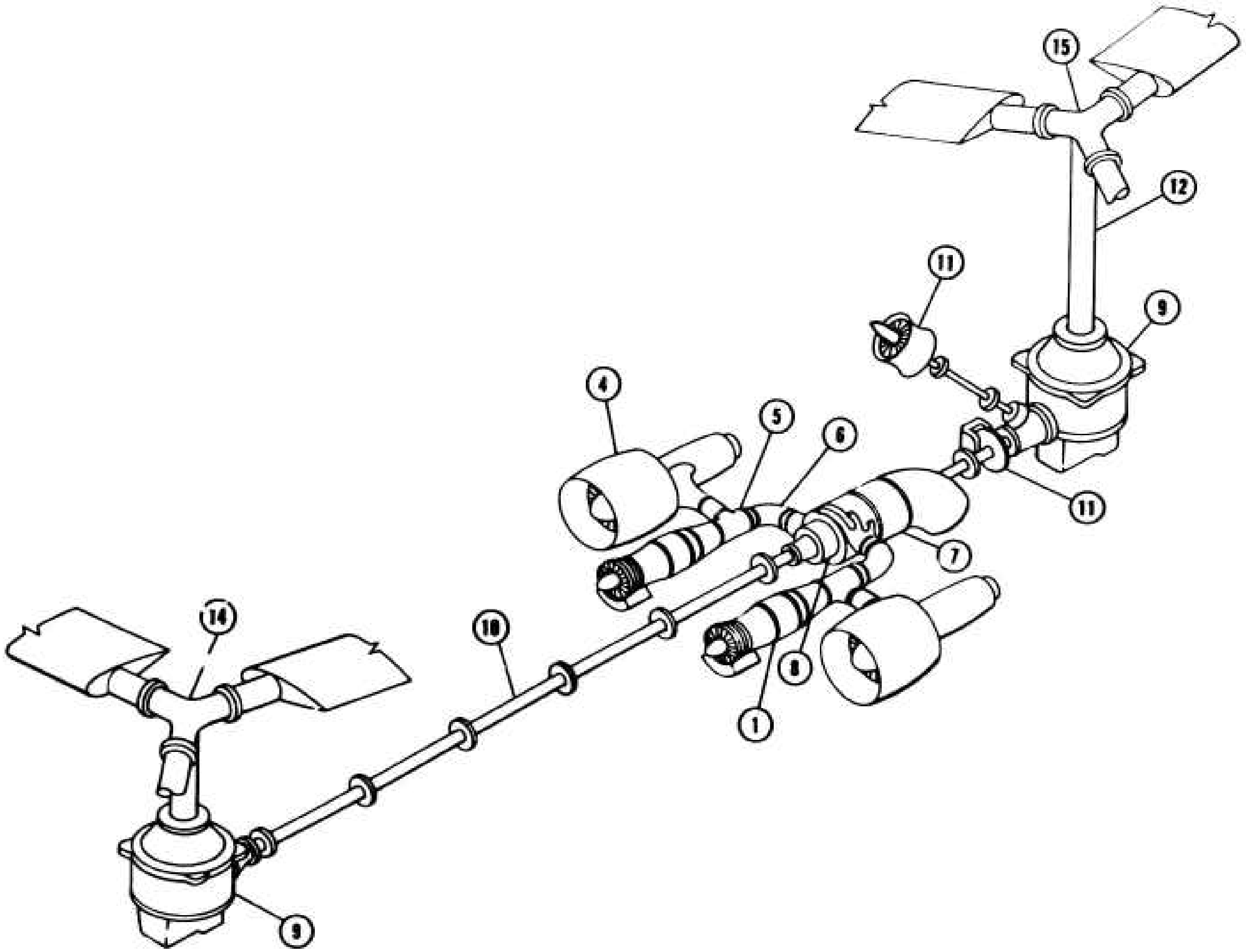


Figure 129. Propulsion System 1b for Tandem Rotor Aircraft

TABLE XX
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 2a - TANDEM ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb) PU Compound	LPU Compound	Composite
		D/L=8 B=9	D/L=11 B=12	D/L=11 B=12
1	Turboshaft engines (2)	540	770	1050
2*	Powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	170	202	201
4	Cruise fan assemblies[2] (2)	300	690	1020
5	Coupling or combining gearbox (1)	189	243	306
6	Decoupling clutch[3] (1)	-	-	148
7	Rotor transmission assemblies (2)	1152	1404	1917
8	Synchronizing shaft assembly and cross-shafting	175	202	238
9	Aft rotor shaft	230	257	354
10	Aircraft accessory drive system, cooling system, rotor brake, and oil for accessory and power drive systems	268	329	445
11*	Engine section	(128)	(312)	(193)
12	Rotor systems[4] (2)	(2300)	(2540)	(4800)
	Total propulsion	3157	4230	5812

*Not shown

1 Includes induction, exhaust, starting

2 Includes fan gearbox, decoupling clutch, variable load device, fan nacelle

3 On composite type aircraft only; omitted on PU & LPU Compound

4 Included here for convenience only; normally part of airframe

Key:
2a - Shaft-driven remote fans
PU - Propulsion-unloaded
LPU - Lift/propulsion - unloaded
D/L - Disc loading
B - Bypass ratio

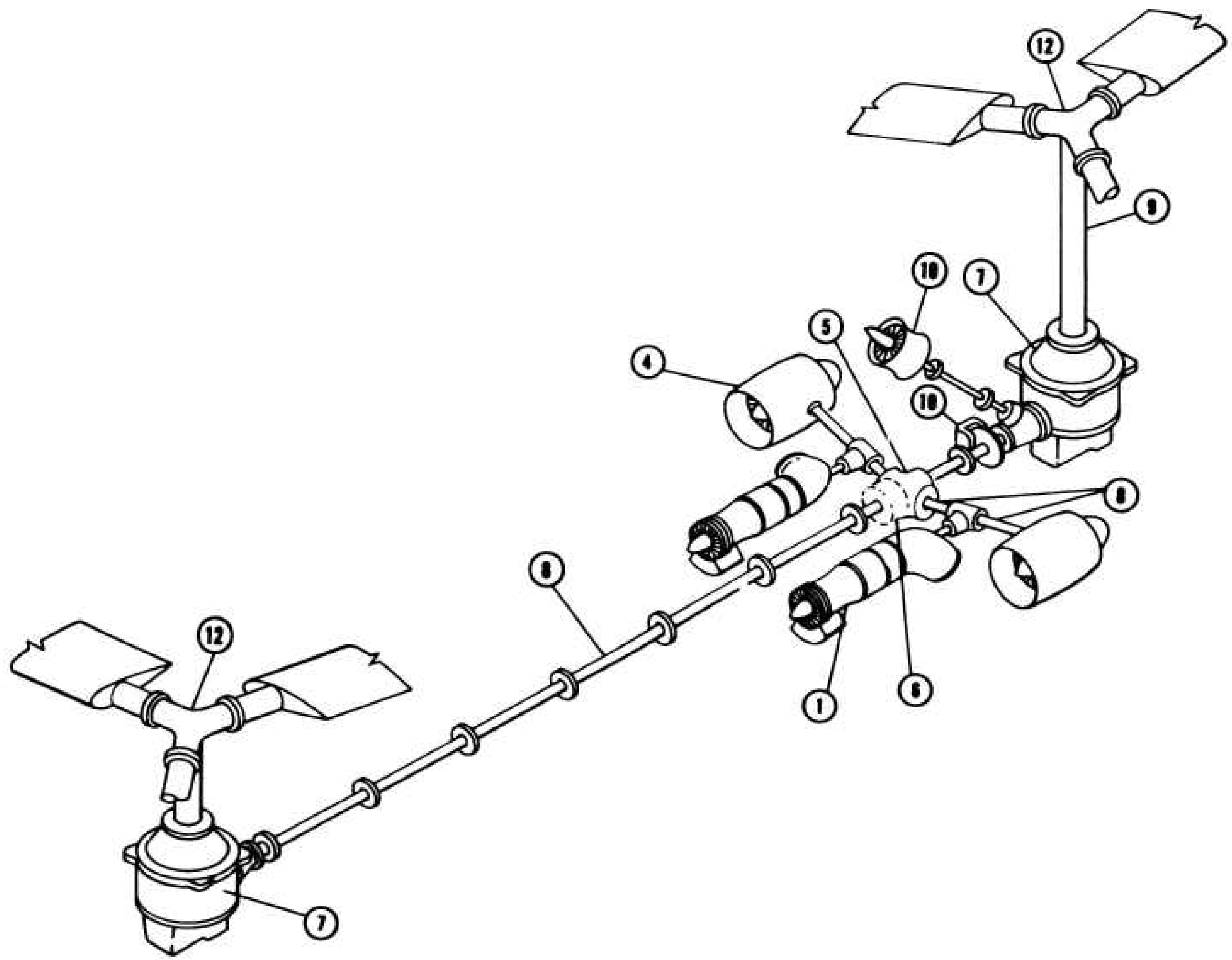


Figure 130. Propulsion System 2a for Tandem Rotor Aircraft

TABLE XXI
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 2b - TANDEM ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=8 B=12	D/L=11 B=9	D/L=11 B=6
1	Convertible shaft turbine fan assemblies (2)	1100	1400	1810
2*	Powerplant subgroups[1] (2)	100	100	100
3*	Fuel system (1)	185	194	215
4	Coupling or combining gearbox (1)	189	230	297
5	Decoupling clutch[2] (1)	-	-	150
6	Rotor transmission assemblies (2)	1188	1341	1845
7	Synchronizing shaft assembly and cross-shafting	176	200	230
8	Aft rotor shaft	234	250	340
9	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	275	314	430
10*	Engine section[3]	(60)	(67)	(75)
11	Rotor systems[3] (2)	(2700)	(2430)	(4740)
	Total propulsion system weight	3447	4029	5417

*Not shown
1 Includes shaft turbine induction system, exhaust, starting
2 On composite type aircraft only; omitted on PU and LPU compounds
3 Included here for convenience only; normally part of airframe

Key:
2b - Convertible cruise fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

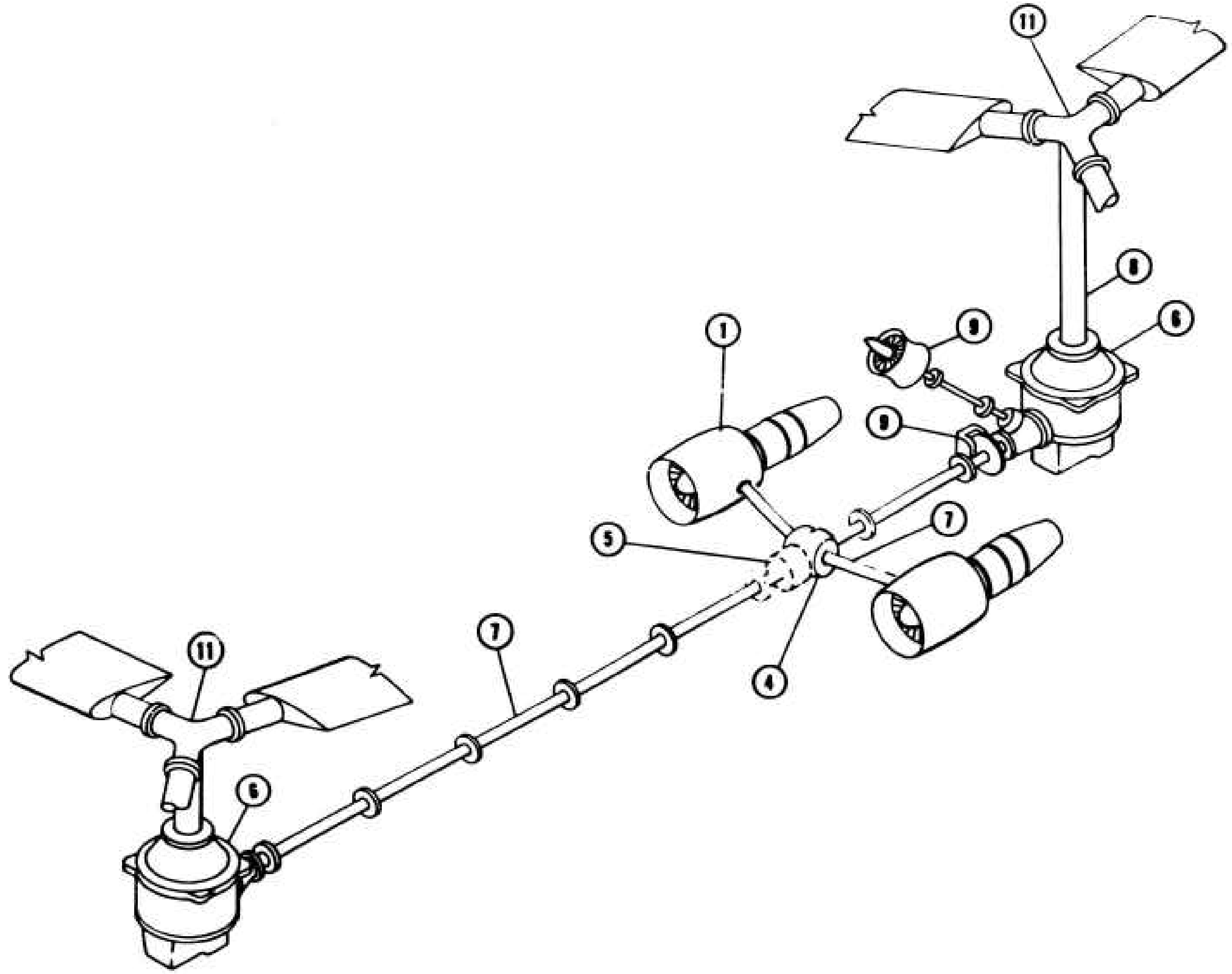


Figure 131. Propulsion System 2b for Tandem Rotor Aircraft

TABLE XXII

PROPULSION SYSTEM WEIGHT BREAKDOWN FOR SYSTEM 3 - TANDEM ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=8 B=9	D/L=8 B=3	D/L=8 B=3
1	Lift system turboshaft engines (2)	1025	835	1070
2*	Lift system powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	338	311	304
4	Cruise fan engines (2)	2780	1230	1640
5*	Cruise system powerplant subgroups[2] (2)	100	100	100
6	Lift system pwrplant bevel gearbox assys (2)	230	173	216
7	Coupling or combining gearbox (1)	301	243	290
8	Synch shaft assy & cross-shafting	235	204	230
9	Aft rotor shaft	455	315	414
10	Rotor t ansmission assemblies (2)	2223	1638	2078
11	A/C access. drive syst,* cooling syst, rotor brake, & oil for access. & power drive systs	550	410	517
12*	Engine section[3]	(223)	(175)	(202)
13	Rotor systems[3] (2)	(7750)	(2870)	(5200)
	Total propulsion system weight	8370	5592	6992

*Not shown

1 Includes lift syst, shaft turbine induction syst, including inlet closure door on composite A/C only, exhausts, & starting

2 Includes cruise syst, shaft turbine induction, exhaust & starting

3 Included here for convenience only; normally part of airframe

Key:
3 - Independent concentric fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

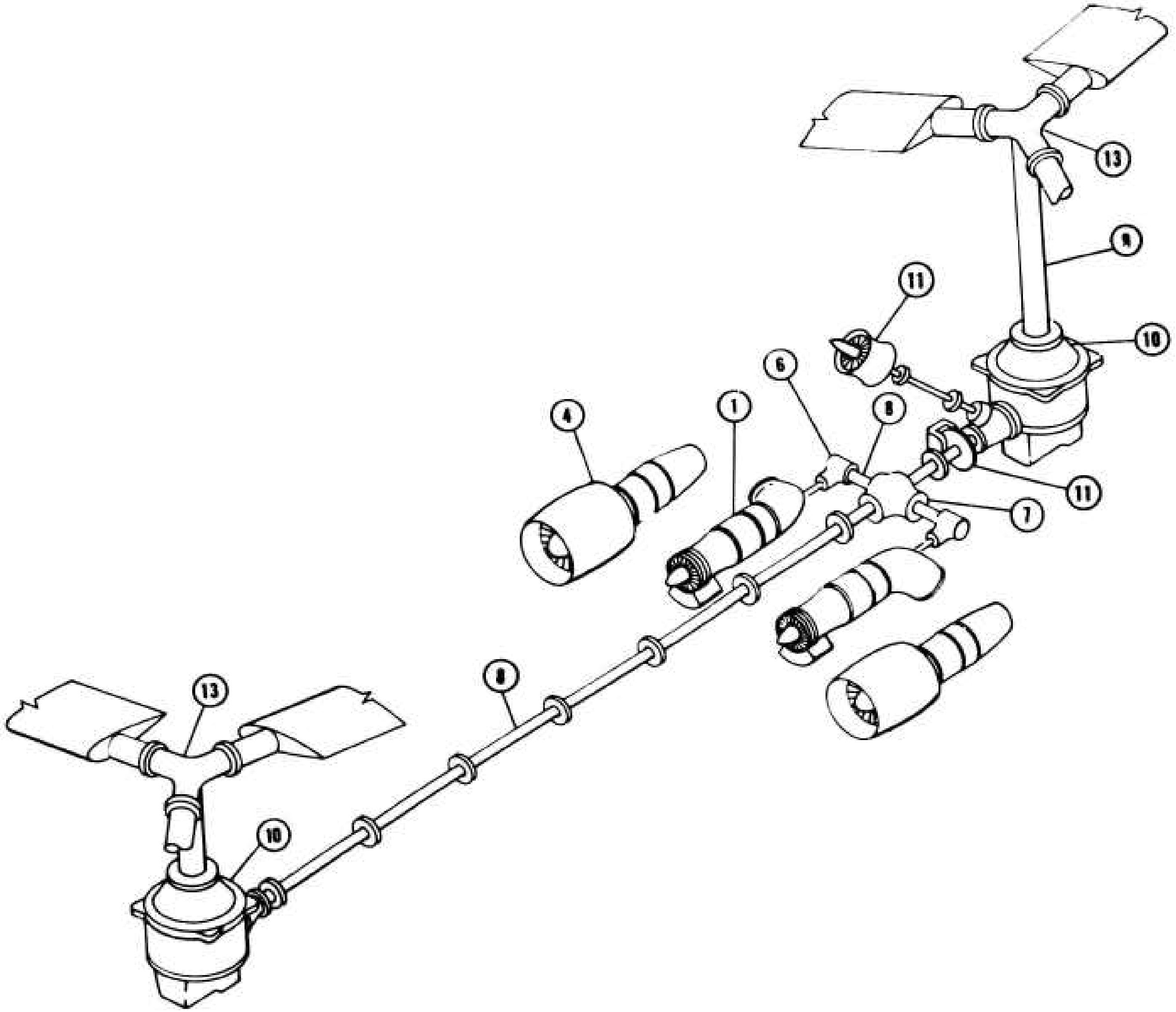


Figure 132. Propulsion System 3 for Tandem Rotor Aircraft

TABLE XXIII
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 1a - SINGLE ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb) PU Compound D/L=8 B=9	LPU Compound D/L=8 B=3	Composite D/L=8 B=3
1	Gas generators	330	416	495
2*	Powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	177	194	195
4	Cruise fan assemblies	340	650	840
5	Diverter valves (2)	60	76	92
6	Gas ducts	33	33	66
7	Remote power turbine assembly[2] (1)	185	235	290
8	Power turbine reduction gearbox (1)	115	120	122
9	Main rotor transmission assembly (1)	1453	1634	1926
10	Main rotor transmission input drive shaft assembly and tail rotor shafting	103	111	130
11	Tail rotor drive system gearboxes	204	255	297
12	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	354	400	468
13*	Engine section[3]	(136)	(165)	(186)
14	Main rotor system[3]	(2556)	(2470)	(4280)
	Total propulsion system weight	3487	4257	5054

*Not shown
1 Includes induction, exhaust, starting
2 Includes variable admission devices
3 Included here for convenience only; normally part of airframe

Key:
1a - Gas-driven tip-turbine fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

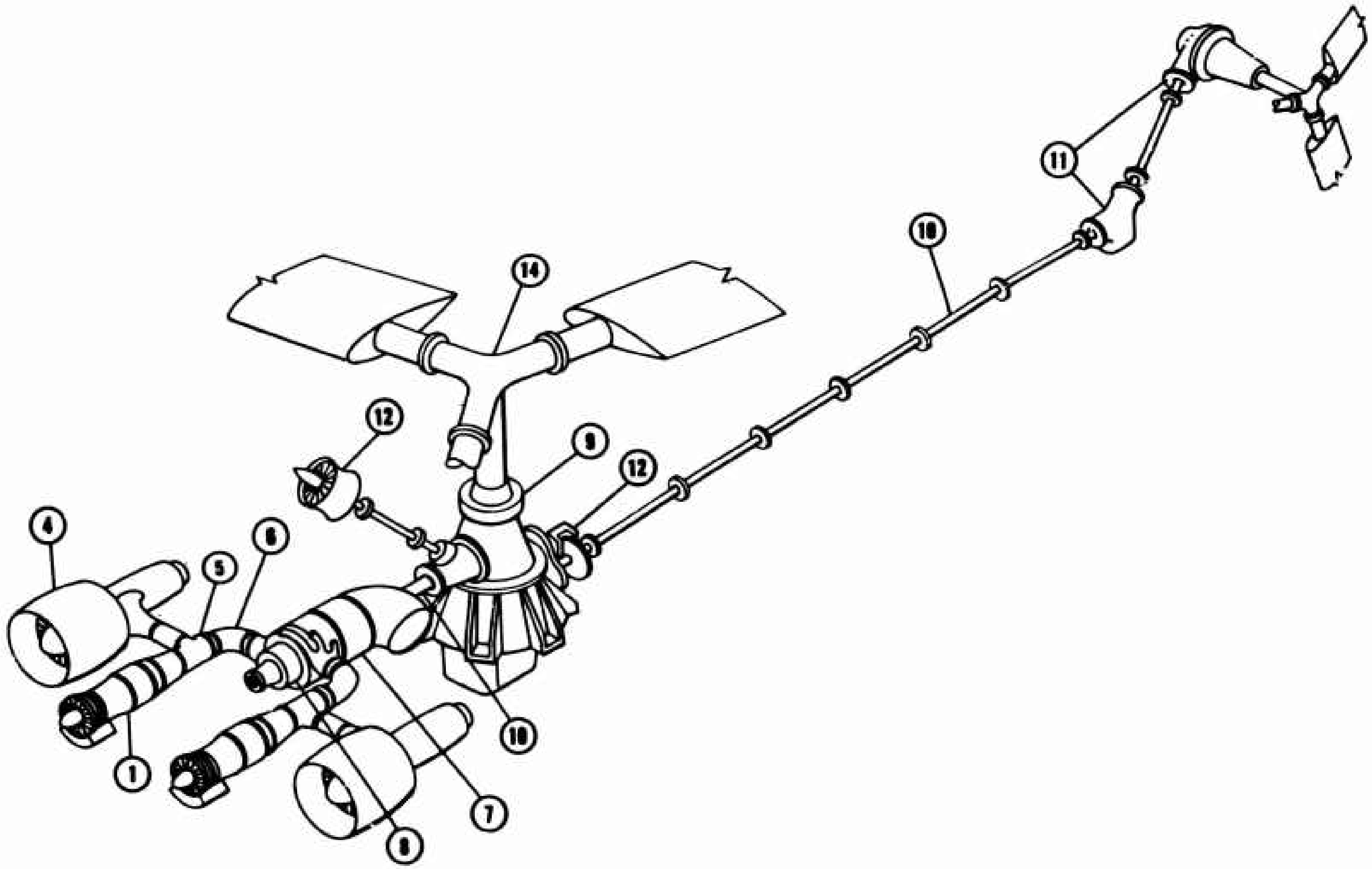


Figure 133. Propulsion System la for Single Rotor Aircraft

TABLE XXIV
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 1b - SINGLE ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=8 B=12	D/L=11 B=12	D/L=11 B=9
1	Gas generators (2)	330	420	510
2*	Powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	175	186	203
4	Cruise fan assemblies[2] (2)	280	1155	1350
5	Diverter valves (2)	60	80	96
6	Gas ducts	33	33	66
7	Remote power turbine assembly[2] (1)	180	237	310
8	Power turbine reduction gearbox (1)	115	120	124
9	Main rotor transmission assembly (1)	1427	1637	2025
10	Main rotor transmission input drive shaft assembly and tail rotor shafting	103	113	135
11	Tail rotor drive system gearboxes	204	257	310
12	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	349	402	492
13*	Engine section[3]	(133)	(179)	(200)
14	Main rotor system[3]	(2556)	(2490)	(4470)
	Total propulsion system weight	3389	4773	5754

*Not shown
1 Includes induction, exhaust, starting
2 Includes variable admission devices
3 Included here for convenience only; normally part of airframe

Key:
1b - Gas-driven hub-turbine fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

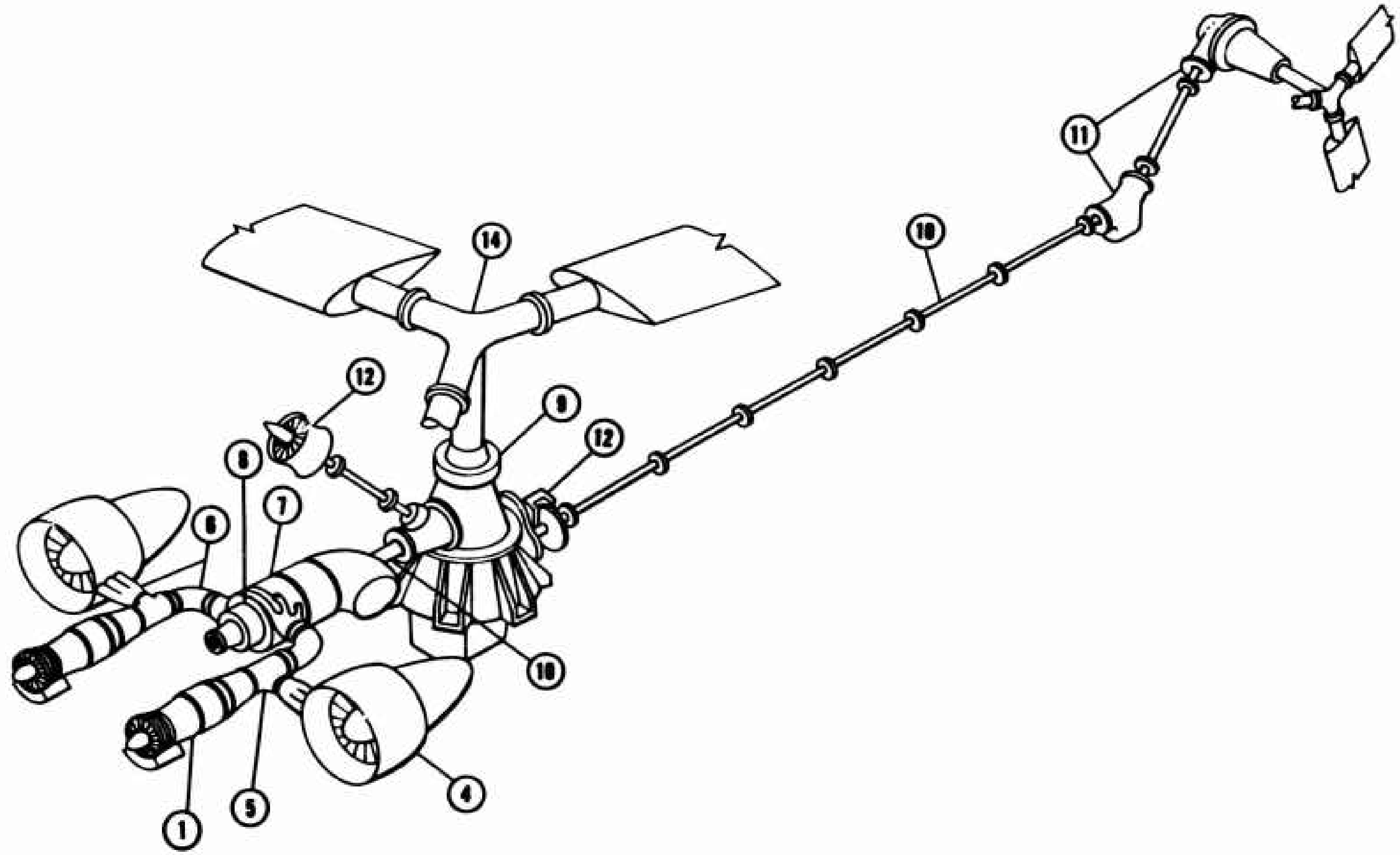


Figure 134. Propulsion System 1b for Single Rotor Aircraft

TABLE XXV

PROPULSION SYSTEM WEIGHT BREAKDOWN FOR SYSTEM 2a - SINGLE ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb) PU Compound D/L=8 B=12	LPU Compound D/L=11 B=12	Composite D/L=11 B=12
1	Turboshaft engines (2)	635	860	1000
2*	Powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	170	210	208
4	Cruise fan assemblies[2] (2)	400	770	980
5	Coupling or combining gearbox (1)	205	264	297
6	Decoupling clutch[3] (1)	-	-	148
7	Main rotor transmission assembly (1)	1404	1728	1940
8	Main rotor transmission input drive shaft assembly, tail rotor shafting and cross-shafting	100	116	135
9	Tail rotor drive system gearboxes	200	265	303
10	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	360	450	506
11*	Engine section[4]	(318)	(173)	(192)
12	Main rotor system[4]	(2463)	(2660)	(4470)
	Total propulsion system weight	3607	4796	5650

* Not shown

1 Includes induction, exhaust, starting, controls

2 Includes fan gearbox, decoupling clutch variable load device, fan nacelle

3 On composite type aircraft only; omitted on PU & LPU compounds

4 Included for convenience only; normally part of airframe

Key:
2a-Shaft drive remote fans
PU-Propulsion-unloaded
LPU-Lift/propulsion-unloaded
D/L-Disc loading
B-Bypass ratio

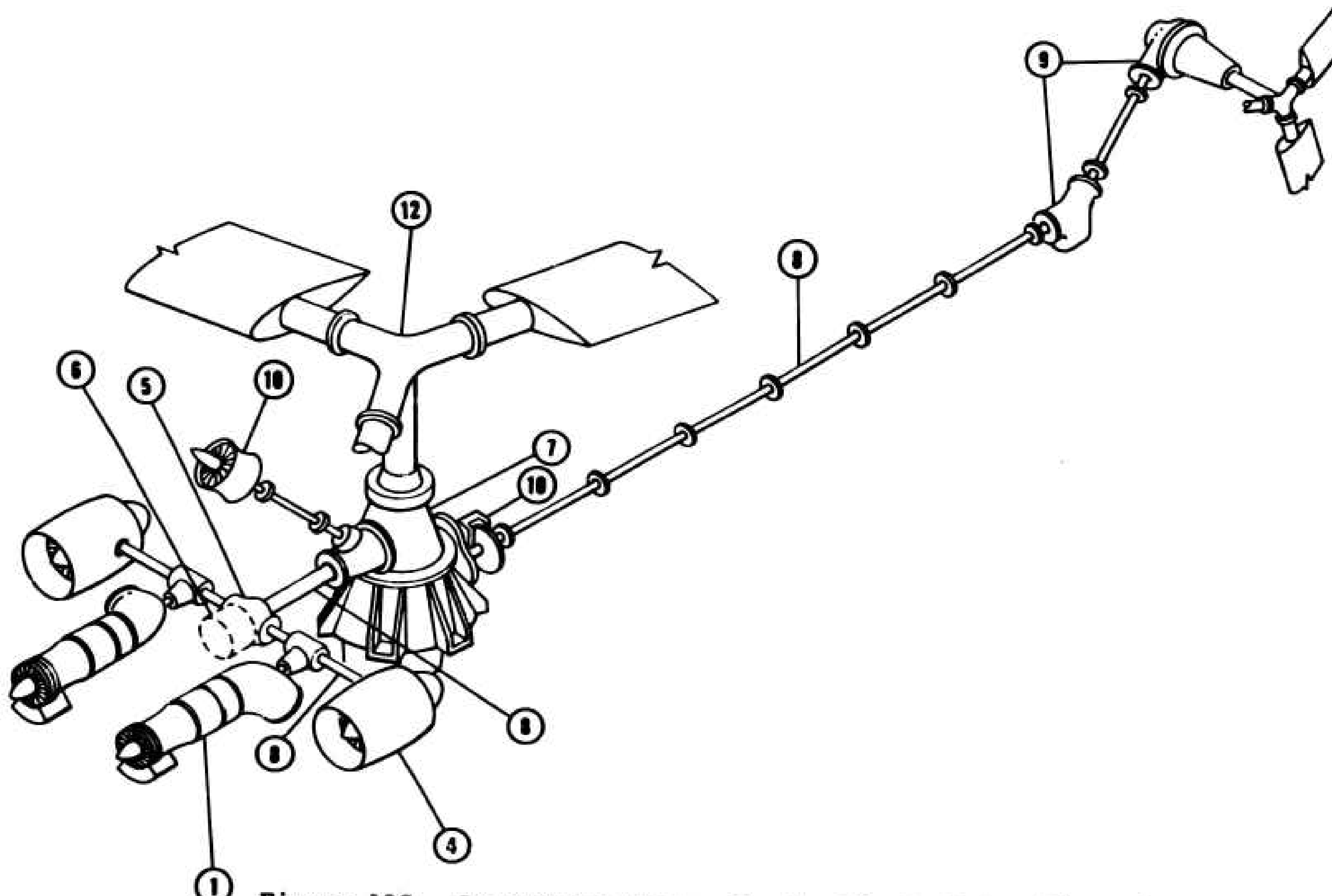


Figure 135. Propulsion System 2a for Single Rotor Aircraft

TABLE XXVI

PROPULSION SYSTEM WEIGHT BREAKDOWN FOR SYSTEM 2b SINGLE ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=8 B=12	D/L=11 B=9	D/L=11 B=6
1	Convertible shaft turbine fan assemblies (2)	1260	1625	1830
2*	Powerplant subgroups[1] (2)	100	100	100
3*	Fuel system (1)	182	211	213
4	Coupling or combining gearbox (1)	207	257	302
5	Decoupling clutch[2] (1)	-	-	148
6	Main rotor transmission assembly (1)	1435	1674	2034
7	Main rotor transmission input drive shaft assembly, tail rotor shafting, and cross-shafting	102	112	135
8	Tail rotor drive system gearboxes	200	260	305
9	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	370	438	530
10*	Engine section[3]	(65)	(70)	(75)
11	Main rotor system[3]	(2603)	(2590)	(4470)
	Total propulsion system weight	3856	4677	5597

*Not shown

1 Includes shaft turbine induction system, exhaust, starting

2 On composite aircraft only; omitted on PU and LPU compounds

3 Included here for convenience only; normally part of airframe

Key:
2b - Convertible cruise fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

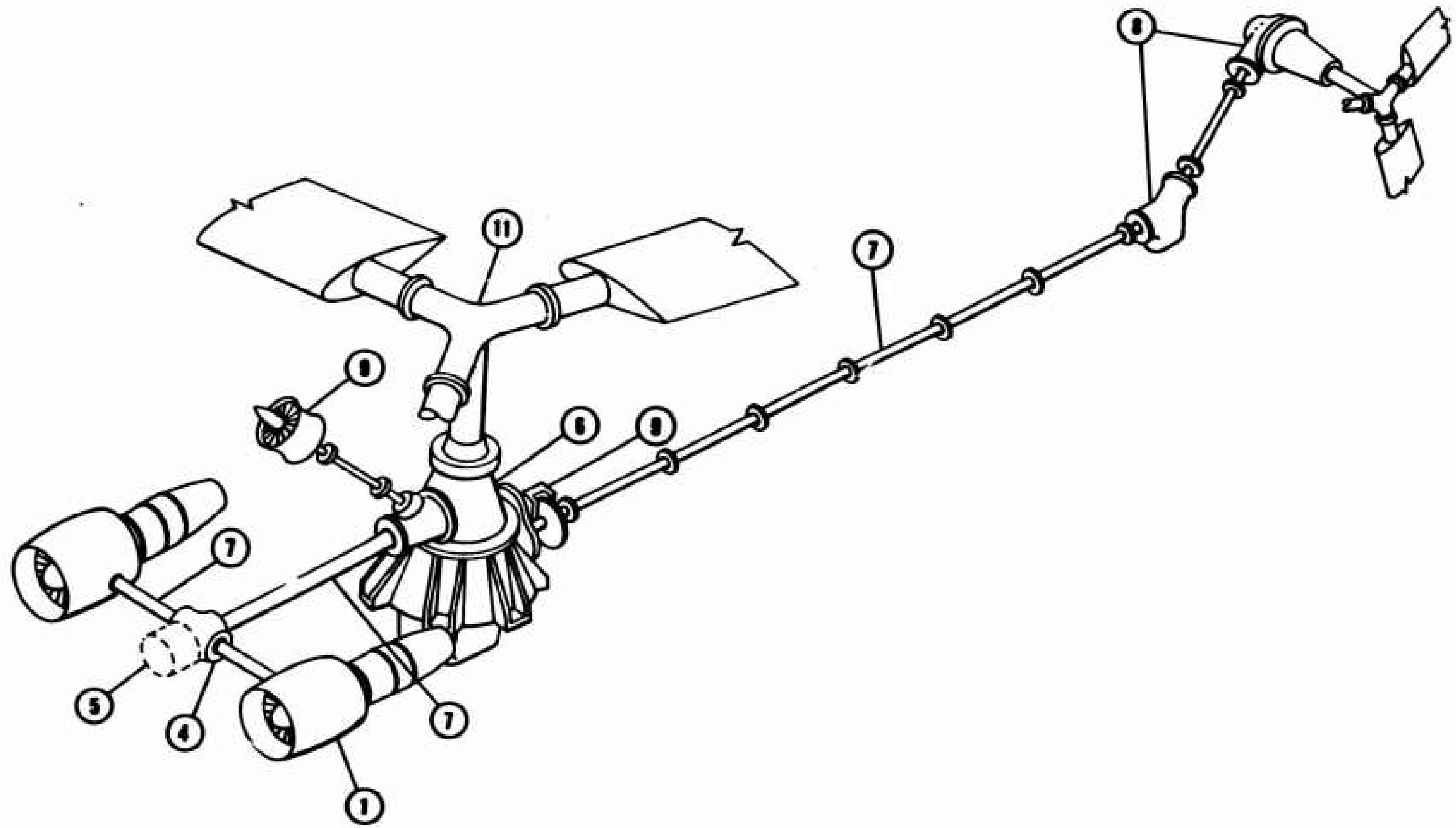


Figure 136. Propulsion System 2b for Single Rotor Aircraft

TABLE XXVII
PROPULSION SYSTEM WEIGHT BREAKDOWN FOR
SYSTEM 3 - SINGLE ROTOR AIRCRAFT

Ref. No.	Item	Weight (lb)		
		PU Compound	LPU Compound	Composite
		D/L=8 B=6	D/L=8 B=3	D/L=8 B=3
1	Lift system turboshaft engines (2)	1000	865	960
2*	Lift system powerplant subgroups[1] (2)	133	133	133
3*	Fuel system (1)	359	305	301
4	Cruise fan engines (2)	2260	1210	1640
5*	Cruise system powerplant subgroups[2]	100	100	100
6	Lift system powerplant bevel gear-box assys (2)	214	166	198
7	Coupling or combining gearbox (1)	288	235	270
8	Main rotor transmission assembly (1)	2340	1764	2196
9	Main rotor transmission input drive shaft assembly, tail rotor shafting, and cross-shafting	156	120	142
10	Tail rotor drive system gearboxes	294	240	279
11	Aircraft accessory drive system,* cooling system, rotor brake, and oil for accessory and power drive systems	627	481	589
12*	Engine section[3]	(212)	(174)	(196)
13	Main rotor system	(7110)	(2680)	(4475)
	Total propulsion system weight	7771	5619	6808

*Not shown
[1]Includes lift system shaft turbine induction system, including inlet closure door on composite aircraft only, exhausts and starting
[2]Includes cruise system shaft turbine induction, exhaust and starting
[3]Included here for convenience only; normally part of airframe

Key:
3 - Independent concentric fans
PU - Propulsion-unloaded
LPU - Lift/propulsion-unloaded
D/L - Disc loading
B - Bypass ratio

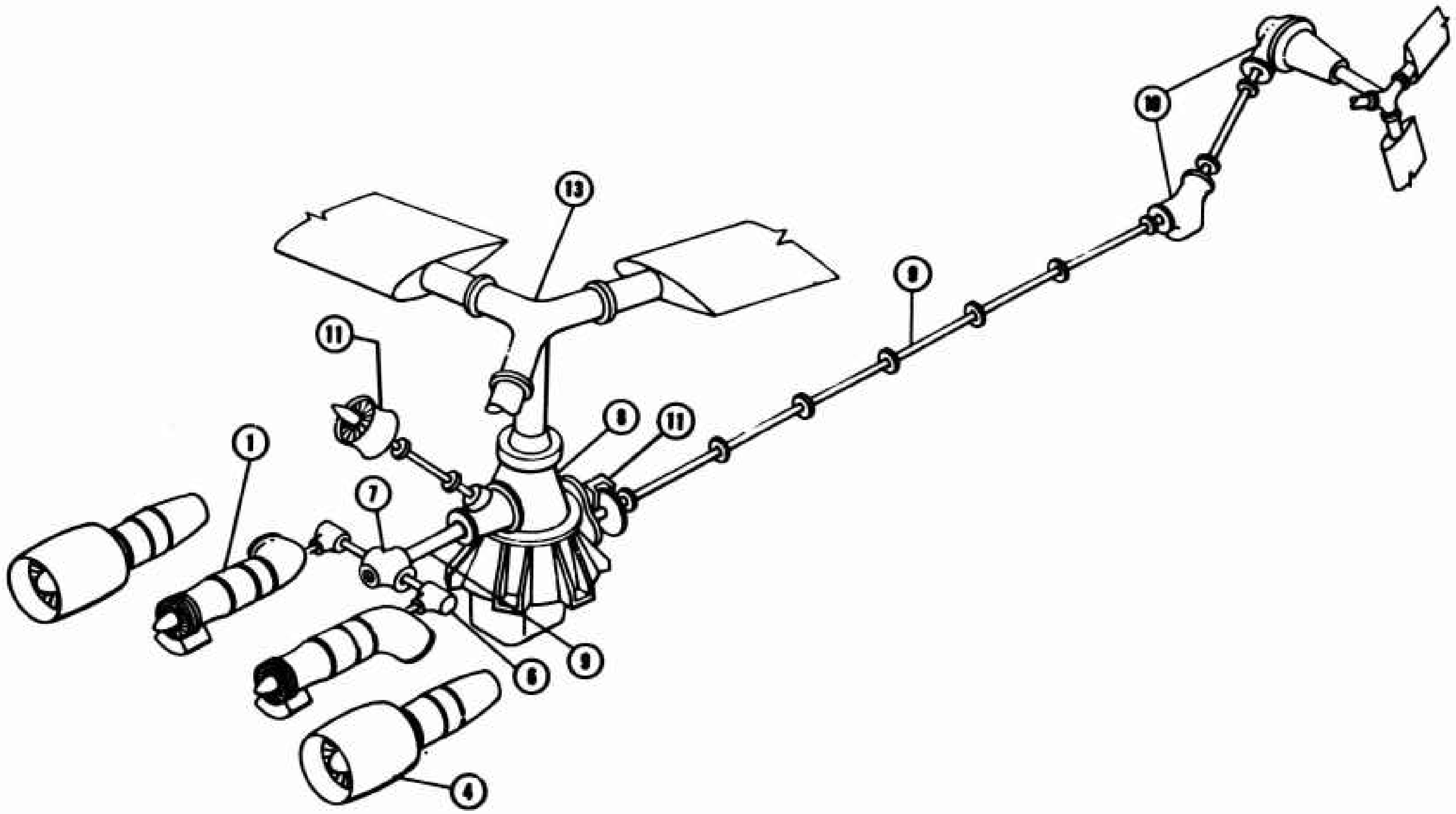


Figure 137. Propulsion System 3 for Single Rotor Aircraft

TABLE XXVIII. PROPULSION SYSTEM AND POWER TRANSFER REQUIREMENTS

		Compound Propulsion			Compound Lift/Propulsion			Composite		
	Performance Requirement	Integ Gas Dr	Integ Shft Dr	Indep	Integ Gas Dr	Integ Shft Dr	Indep	Integ Gas Dr	Integ Shaft Dr	Indep
1R	Rotor shaft power range	60-100%	50-100%	24-100%	0-100%	0-100%	0-100%	0-100%	0-100%	0-100%
2R	Rotor shaft rpm range	88-100%	90-100%	75-100%	64-100%	60-100%	76-100%	0-100%	0-100%	0-100%
3R	Rotor overrunning clutch	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes
4R	Rotor decoupling	No	No	No	No	No	No	Yes	Yes	Yes
5R	Rotor cruise power (opt)	75%	75%	0	0	0	0	0	0	0
1F	Fan thrust range	0-100%	0-100%	0-100%	0-100%	0-100%	0-100%	0-100%	0-100%	0-100%
2F	Fan rpm range	0-100%	90-100%	90-100%	0-100%	60-100%	90-100%	0-100%	90-100%	90-100%
3F	Fan clutch	No	Yes	No	No	Yes	No	No	Yes	No
1	Variable fan inlet guide vanes	No	Yes	No	No	Yes	No	No	Yes	No
2	Variable fan turbine arches	Yes	No	No	Yes	No	No	Yes	No	No
3	Variable power turbine arches	Yes	No	No	Yes	No	No	Yes	No	No
4	Diverter valve	Yes	No	No	Yes	No	No	Yes	No	No

POWER TRANSFER SYSTEMS

REQUIREMENTS FOR POWER TRANSFER DEVICES

In the compound/composite aircraft configuration analysis curves were prepared giving rotor power and auxiliary thrust required for the speed range of each system. These curves define a set of power management requirements. The development of an efficient power transfer system to meet these requirements is then the key to the successful use of cruise fan powerplants for compound and composite aircraft. Obviously, for the independent cruise fan propulsion system there is no need for special power transfer devices since this control can be gained through the throttle and rpm control of both the cruise fan and the shaft turbine. The tradeoff then is between the added weight of the independent system and the weight, complexity, and efficiency of the power transfer devices in the integrated systems.

Figures 60 through 69 give some indication of the interconnection requirements of the five propulsion systems coupled with the single and tandem rotor airframe configurations. Referring to these figures and the previous airframe/propulsion system discussion on the basic propulsion system requirements, a set of general requirements for a power transfer system can be evolved. If this system is to be applicable for all compound and composite aircraft, it must meet the following requirements:

1. Permit maximum rotor power during hovering with little (or zero) loss due to fan or system.

2. Permit maximum thrust during cruise flight with little (or zero) loss due to power takeoff or system.

3. Permit gas generator power to be used efficiently in any rotor/fan combination at all speeds within engine capability, and provide adequate margin of power available through transition.

4. Permit reduction in rotor rpm with minimum degradation of fan thrust during cruise of a compound configuration.

5. Provide simple, constant speed control system (selectable rotor drive/fan rpm) with automatic power control of the

gas producer to provide the torque-on-demand from rotor drive and/or fan drive.

6. Provide for decoupling rotor drive for composite aircraft configuration.

7. Provide control system for both rotor and fan throughout flight regime from hover to maximum speed cruise.

The results of the matching study indicate more specific requirements regarding rotor and fan power. Table XXVIII summarizes these requirements and is derived from the resultant hover and cruise conditions to which the optimum aircraft have been designed. An analysis of the various methods of satisfying these requirements follows. Analysis of the efficiency, complexity, and feasibility of each system have led to the choice of variable inlet guide vanes for fan thrust control in the shaft drive systems and variable fan and power turbine arches for fan and rotor power control in the gas-drive systems.

POWER TRANSFER METHODS

Gas-Drive Systems

Power absorption of the rotor is controlled by blade pitch and power absorption of the fan is controlled by fan speed. The problem remaining is the management of the power transfer to these systems in the form of high-energy hot-gas exhaust from the gas generators. The flapper valve in both branches of the diverter valve and the rotor brake permit hot gas flow to the remote power turbine to be diverted and the rotor to be stopped and stowed in the composite case. Methods of continuous flow modulation to the remote power turbine and to the gas-driven fans were studied, and the following are the results of this analysis.

The diverter valve was analyzed as a possible flow modulating device to regulate the division of gas flow between the cruise fans and the remote power turbine in the gas-driven systems. However, data from Reference 3 plotted in Figure 138 demonstrated that the diverter valve would be a very inefficient means of regulating fan thrust and turbine power, because of the large pressure losses for partial valve opening. As a consequence, the diverter valve became just an open-and-close control device to direct the gas generator exhaust gas to the power turbine for the hover mode, to permit flow to both power

turbine and fans for the compound aircraft in the cruise mode, and to eliminate power turbine flow and permit only fan flow for the composite aircraft in the cruise mode. The result was that the diverter valve satisfied requirements for zero fan losses in hovering and zero power turbine losses during the cruise mode of the composite aircraft.

To accomplish the required flow modulation for the gas-driven systems, variable admission arches, variable turbine stator vanes, and variable turbine exhaust nozzles were all considered for both the remote power turbine and the fan turbine. Any of these flow modulation devices on the remote power turbine, in conjunction with rotor collective pitch control, permit constant rotor speed operation at any power setting. The remote gas-coupled cruise fan was envisioned as a variable speed powerplant with fan speed a function of the fan input power and thus controlled by the flow modulation control on the fan turbines.

In the variable admission arch configuration, the turbine inlet annulus was divided into two separate segments, each one ducted separately and the flow controlled by separate gate valves. The losses in a partial admission system have been discussed in References 2, 12, and 24, including interference losses at either end of the arc of admission, and pumping and windage losses in the inactive arc of the rotor annulus.

The effect of partial admission on turbine efficiency and power from these references was plotted in Figure 139. The curve from the efficiency correlation in Reference 24, which was the most optimistic of the three, produced approximately the same power correlation as the ideal curve of Figure 138, which indicated those results to be optimistic, also. The effect of the number of variable admission arches was apparent from Figure 138, which indicated that increasing the number would result in a closer approximation of the ideal curve.

Consequently, it would appear that the power transfer system for the compound aircraft, for instance, could be designed with the proper number of variable admission arches so that one of the intersections of the actual power curve with the ideal would correspond exactly to the cruise gas flow. Figure 140 shows the effect on fan thrust of many variable admission arches at the fan turbine inlet, closely approximating the ideal fan thrust. Efficient use of various combinations of

shaft power and fan thrust would require a large number of variable admission arches.

Also plotted on Figure 140 is an approximation of the variable fan rpm as a function of the fan turbine flow, illustrating that the gas-coupled fan was a variable speed powerplant.

The amount of performance data for variable turbine stator vane configurations was very limited, since this device has been incorporated by only one manufacturer in an engine under development for the U.S. Army Tank Automotive Center. The problems of variable stator vanes would include distortion of parts at high temperatures which makes stator vane turning and accurate control difficult, and leakage of the flow around the stator vanes and its effect on performance. The source of the losses at part power for a turbine with variable stator vanes was demonstrated qualitatively with the turbine velocity diagrams in Figure 141.

Closing the stator vanes at part power resulted in significant incidence angles at the inlet to the rotor blade and incidence losses, as well as in increased rotor exit tangential velocity and exit kinetic energy losses. From Reference 8, the limited data on variable stator vane performance was plotted in Figure 142, illustrating that the turbine efficiency could be maintained at a reasonable level over a wide range of flow. However, the performance of the turbines with variable stator vanes was not as efficient at part power operating points as that of the turbines with variable admission arches.

Again considering the velocity triangles in Figure 141, the reduction in flow at part power operation for the turbines with variable area exhaust nozzles was accomplished by reducing the expansion through the stage and reducing the stage work. The result was incidence angles and incidence losses at the rotor blade inlet and losses due to rotor exit tangential velocity. Also, increased pressure ratio across the exhaust nozzle produced increased exhaust kinetic energy losses. The additional losses involved in this system disqualified it from further consideration for the remote power turbine.

Shaft-Driven Systems

As in the gas-driven systems, power absorption in the rotor of shaft-driven systems is controlled by blade pitch. Since the fan is directly coupled to the rotor shaft and must run at a

constant set speed unless declutched, the problem remaining is the method of modulating the power absorption of the fan system. In these systems the rotor decoupling for the composite aircraft configurations is accomplished by a decoupling clutch.

Among the power transfer methods analyzed for the shaft-driven fan systems were variable fan inlet guide vanes, variable stagger (or variable pitch) fan rotor blades, and variable fan inlet diaphragms. Variable inlet guide vanes were proposed for the compound turbofan engine in Reference 16, which is referred to as a convertible engine in this study. In the cruise mode of operation, with the clutch engaged between the fan-turbine low-pressure compressor shaft and the fan, the variable inlet guide vanes were to modulate the flow through the fan and to provide an infinitely variable thrust-shaft horsepower relationship. This relationship is shown in Figure 143 and was taken from Reference 16.

Calculations were made for a convertible fan system with variable inlet guide vanes, with cascade losses assumed to be proportional to the cascade relative kinetic energy and suitable incidence losses. The resulting thrust-power relationship closely approximated the manufacturer's data in Reference 16 and is presented in Figure 143. The fan rotor and stator incidence angles and velocity ratios for off-design flows were plotted in Figure 144. Because of the poor aerodynamic conditions in the stator at the lower fan flows, indications were that surge would occur at low fan flows. Reference 23 confirmed that incidence angles increase significantly at lower flows with variable inlet guide vanes. This problem of fan surge could compromise the performance of the convertible fan at low fan thrust. Tests are being conducted by the engine manufacturer to prove this concept.

Another power transfer method was the variable stagger (or variable pitch) fan rotor blade. With the lack of any performance data from the engine manufacturers, calculations were made with cascade loss and incidence loss assumptions similar to those made for the convertible fan calculations. The results of these calculations were not satisfactory, however, and resulted in an improbable thrust-power relationship. However, the fan rotor and stator incidence angles and velocity ratios presented in Figure 145 again indicated probable surge problems at low fan flows.

A third power transfer system for possible use with the shaft-driven fans was the variable fan inlet diaphragm, a device for throttling the inlet flow to the fan. References 9 and 15 discussed some of the NACA work with inlet annulus blockage in multistage compressors, where inlet baffles produced a substantial shift in the surge line of the compressor and improved the low-speed surge margin. Considerable development effort was devoted to a snap-ring inlet baffle for the same purpose. Performance calculations were made for the variable fan inlet diaphragm, with loss assumptions which included losses due to contraction of the flow at the inlet baffle and subsequent expansion in the rotor, windage losses in the inactive part of the rotor annulus flow passage, cascade losses proportional to the average kinetic energy, and rotor and stator incidence losses. The resulting thrust-shaft horsepower relationship, shown in Figure 146, was definitely poorer than that for the variable inlet guide vanes.

On the basis of these results, use of the variable fan inlet diaphragm was disqualified from further consideration among the power transfer methods for shaft-driven fans. Of the remaining methods, the variable inlet guide vanes seemed to be the most promising, although it appeared that variable fan stator vanes must be tied in with such a system to prevent vane stall and to eliminate the surge problem. The complexity of this system should be less than that of variable-stagger fan rotor blades.

Selected Systems

Performance - As indicated in the previous paragraphs, the method selected for power modulation to the remote power turbine and gas-driven fans was use of variable admission arches. Partial admission information is not new, and it is believed the flow modulation, although somewhat complex, can be handled throughout the power range with minimum losses.

The method selected for shaft-driven fans has not, however, been completely tested. Variable inlet guide vane fans have been used for small-range modulation, but as yet they have not been tested throughout the thrust range. Further work in the 0- to 50-percent thrust range is required to obtain answers to questions regarding fan stall and thrust output in this area.

Control Requirements - The basic functions of a propulsion control system are to provide the pilot with means of starting and stopping propulsive units and controlling their speed and

power output. This must be accomplished in a manner allowing the aircraft to perform properly through its intended flight envelope without exceeding propulsion component design limits or acceptable pilot workloads.

In order to give an illustration of the basic control requirements both of integrated shaft-driven or gas-coupled systems and of independent cruise fan propulsion systems, simple functional schematic diagrams are presented in this section. In each case the diagrams are intended to cover all three aircraft types considered in this report. For the sake of simplicity, single-engine systems are depicted; however, the control functions involved are conceptually similar to those required for twin-engine aircraft.

Figure 147 shows a potential control system for an integrated gas-coupled propulsion arrangement; also, component control functions are described briefly. A conventional means of rotor system speed and power control is indicated. Additional control elements are added to provide the required cruise fan control and relative power management functions.

Figure 148 shows basic elements of a possible control system for an integrated shaft-coupled propulsion configuration; a brief description is also presented. Again, rotor control is conventional. Means for fan control and system power management also are indicated.

Figure 149 shows an equivalent system for the independent cruise fan application. Means for control of both lift and cruise subsystems are presented.

POWER TRANSFER SYSTEM DESIGN, DEVELOPMENT, AND PRODUCTION RESPONSIBILITY

In the development of the cruise fan propulsion systems for the compound and composite aircraft, the interface between the airframe contractor and the engine contractor becomes of primary importance because of the power transfer components necessary to the integration of the propulsion system. Analysis of the propulsion systems indicates that a division of prime contract responsibility can be made as a function of each system, as follows:

Gas-Coupled System (Tip Turbine and Axial Turbine)

Due to the interrelationship between the gas generator, remote cruise fan, power turbine, and control system, the complete power package would be supplied and qualified as a unit by the engine manufacturer, with space provisions and mounting requirements supplied by the airframe manufacturer.

Shaft-Coupled (Remote Cruise Fan)

Other than the basic shaft turbine, all elements would be the prime responsibility of the airframe contractor. Subcontracting of many components (such as the control system, fans, and gearboxes) to engine or propeller manufacturers would be likely.

Shaft-Coupled (Convertible Cruise Fans)

Since the primary power transfer system is an integral part of the convertible cruise fan, the engine manufacturer would be prime contractor. Other systems (such as controls, decoupling gearboxes, etc.) would remain the prime responsibility of the airframe contractors.

Independent System

Normal division of prime responsibility would be followed, since there is no power transfer system.

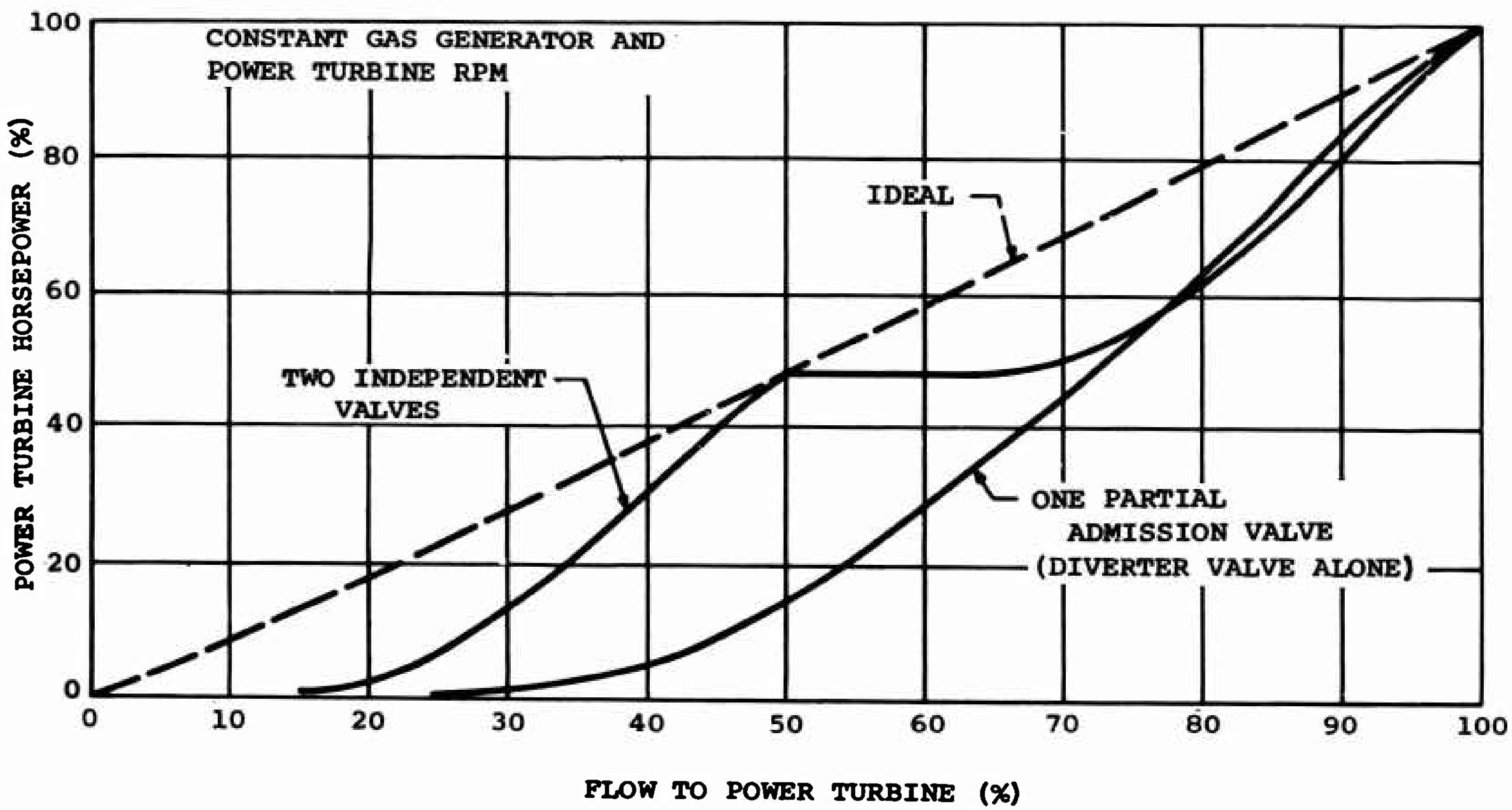


Figure 138. Power Turbine Horsepower With Partial Admission Throttling Valves

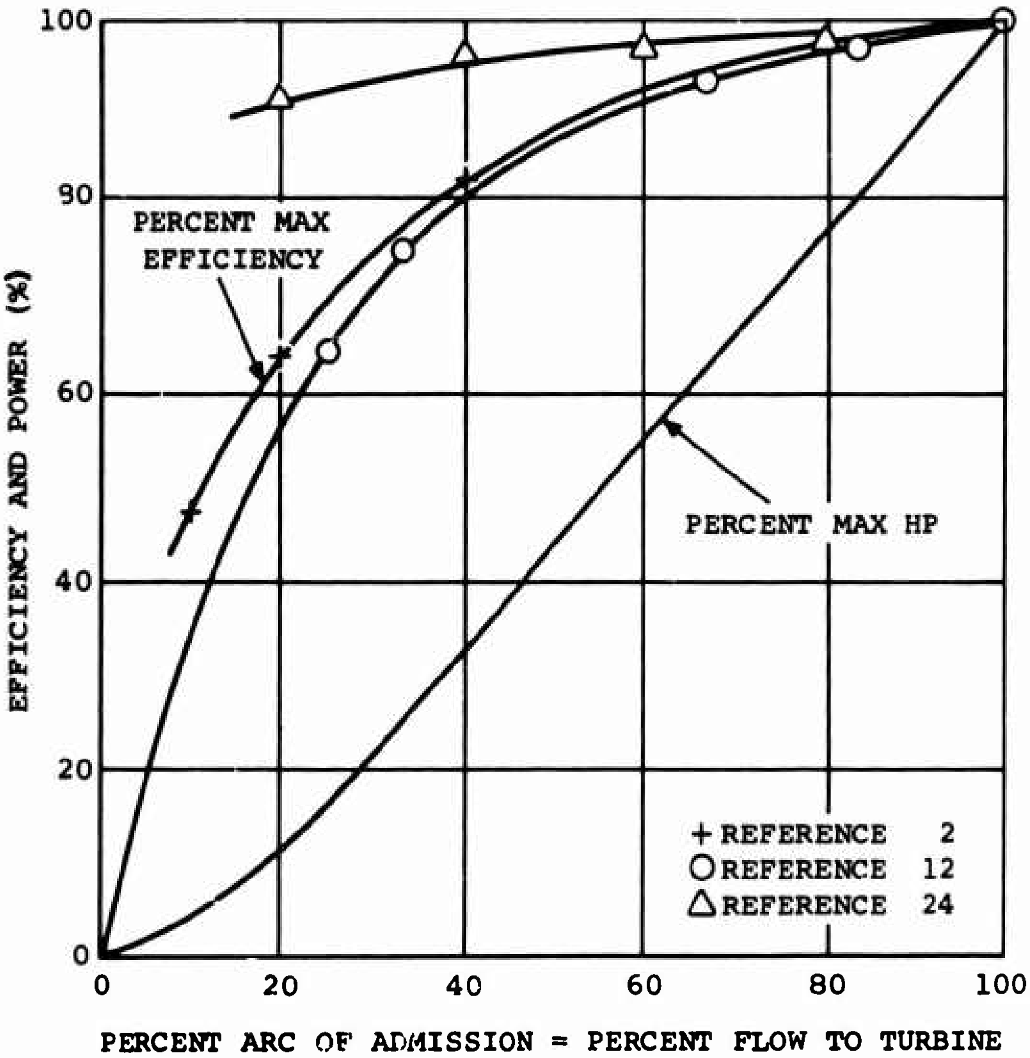


Figure 139. Partial Admission Turbine Efficiency and Power

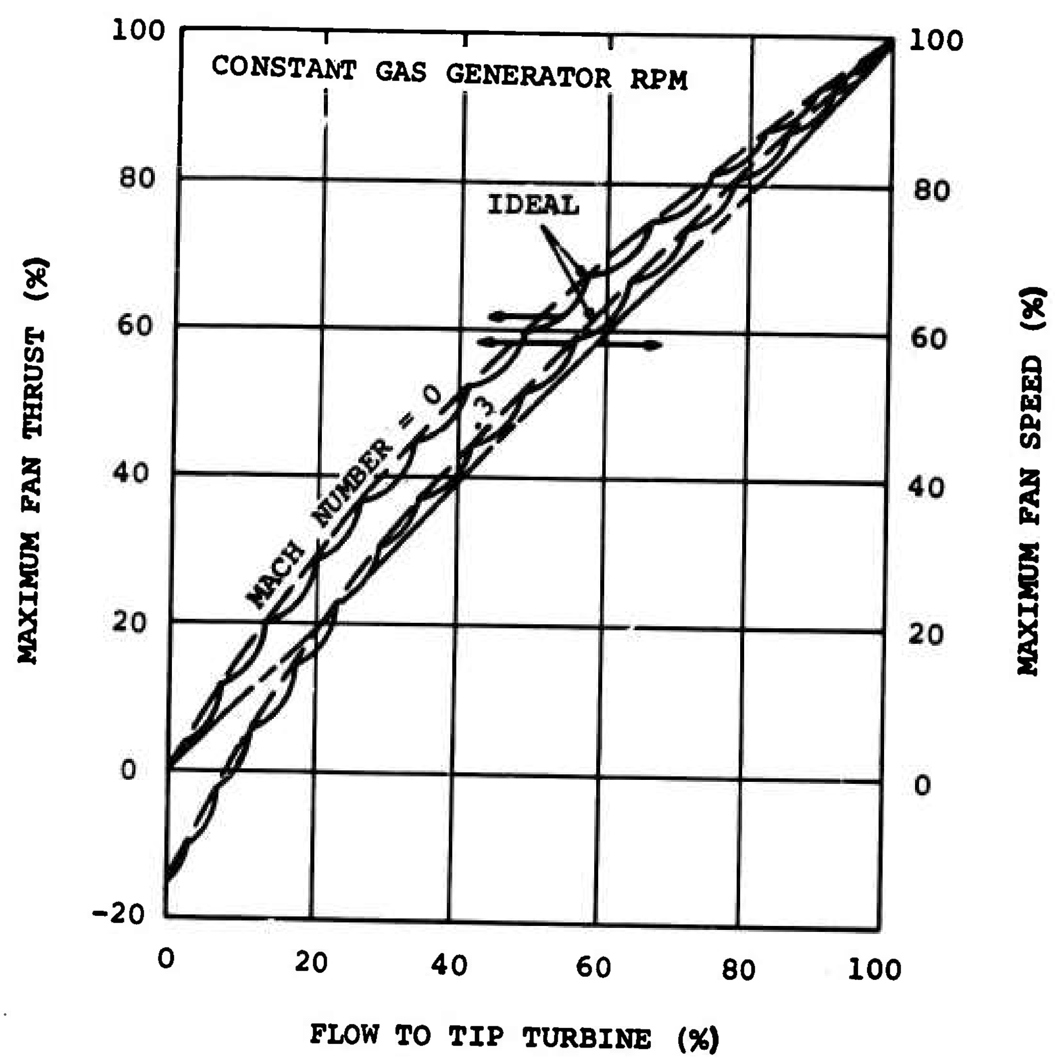


Figure 140. Partial Admission Tip-Turbine Fan Thrust and Speed

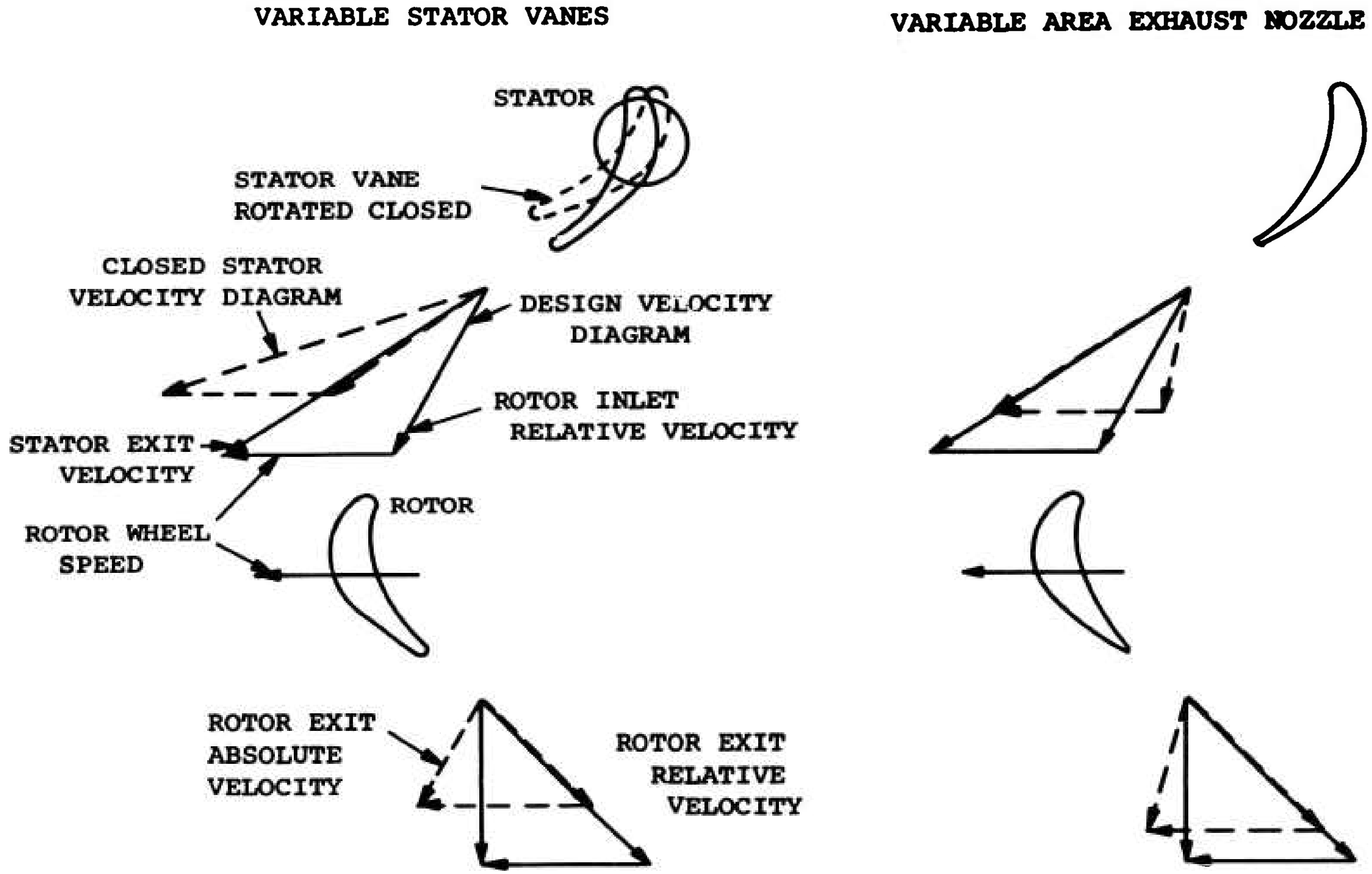


Figure 141. Turbine Velocity Diagrams for Variable Turbine Geometry

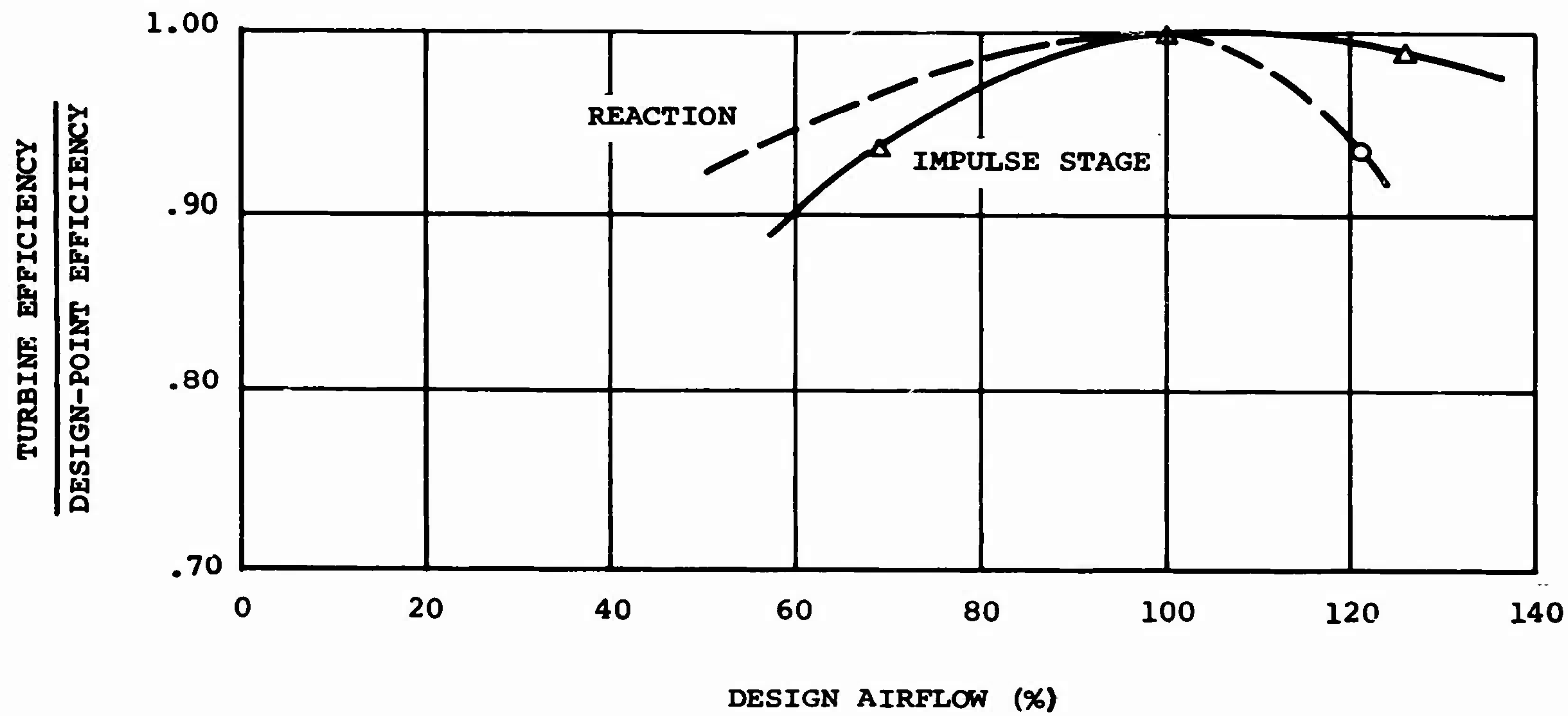


Figure 142. Turbine Performance With Variable Stator Vanes

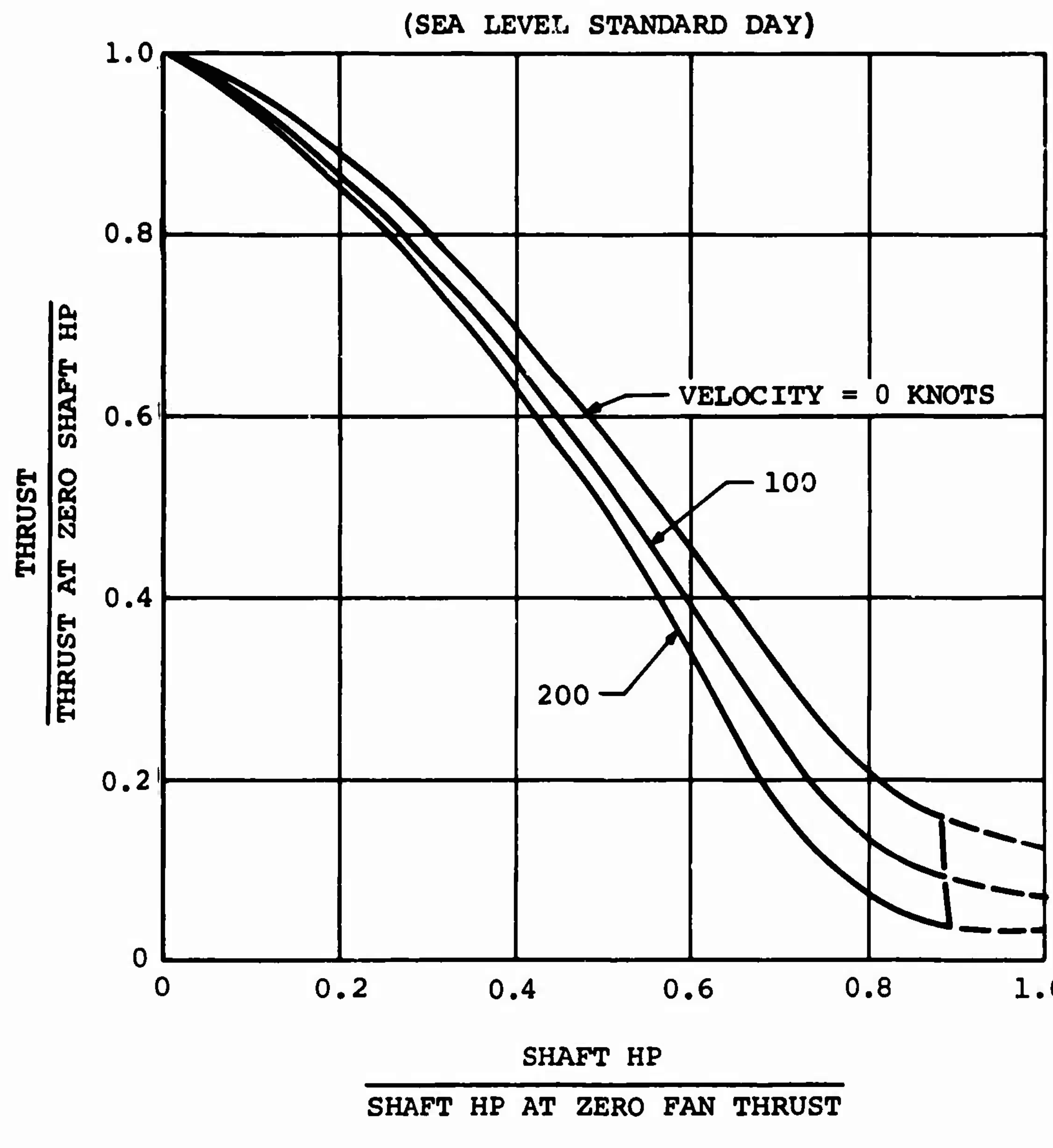


Figure 143. Generalized Thrust-Horsepower Relationship for Convertible Cruise Fan

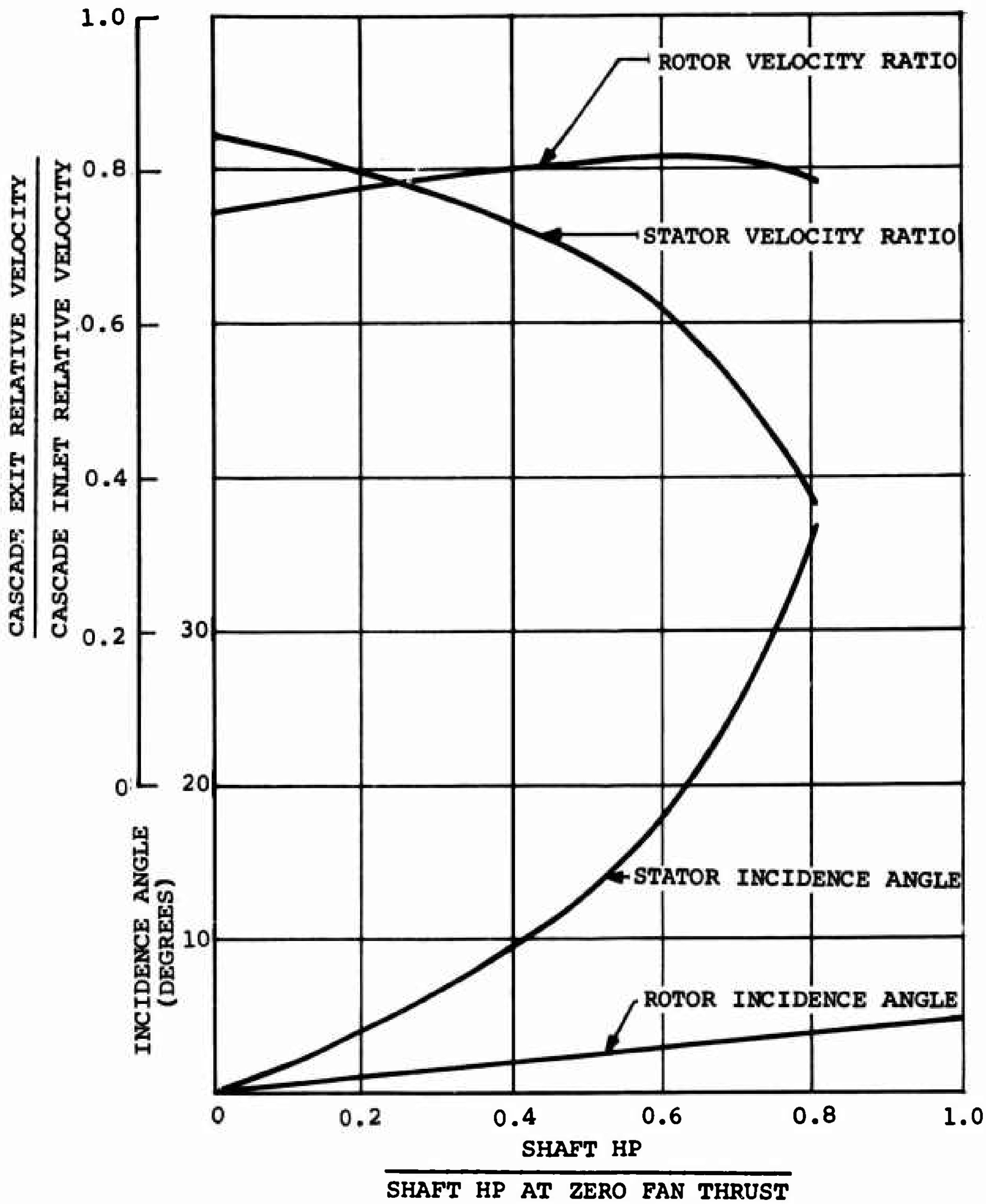


Figure 144. Incidence Angles and Velocities for Fan With Variable Inlet Guide Vanes

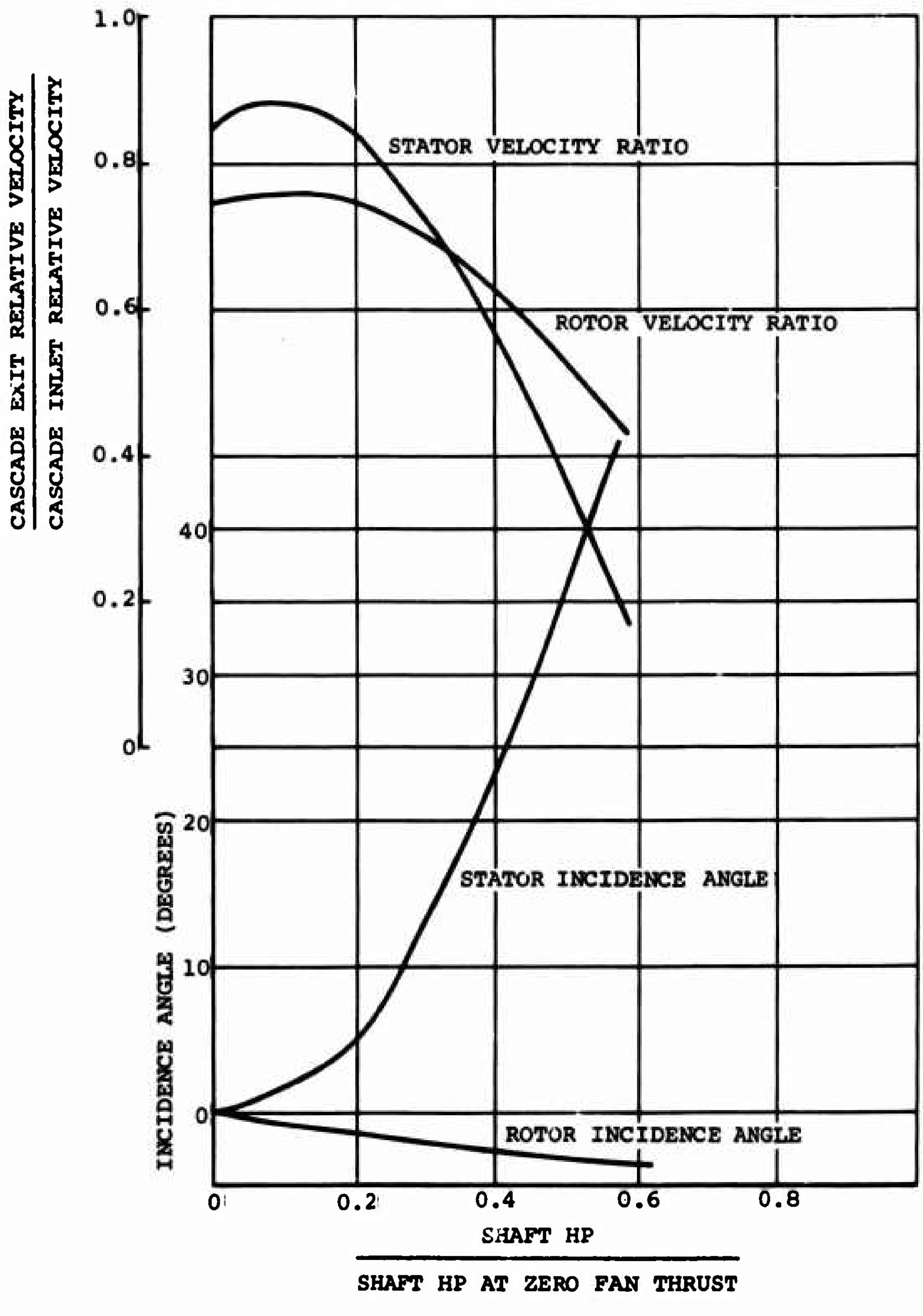


Figure 145. Incidence Angles and Velocities for Fan With Variable Stagger Rotor Blades

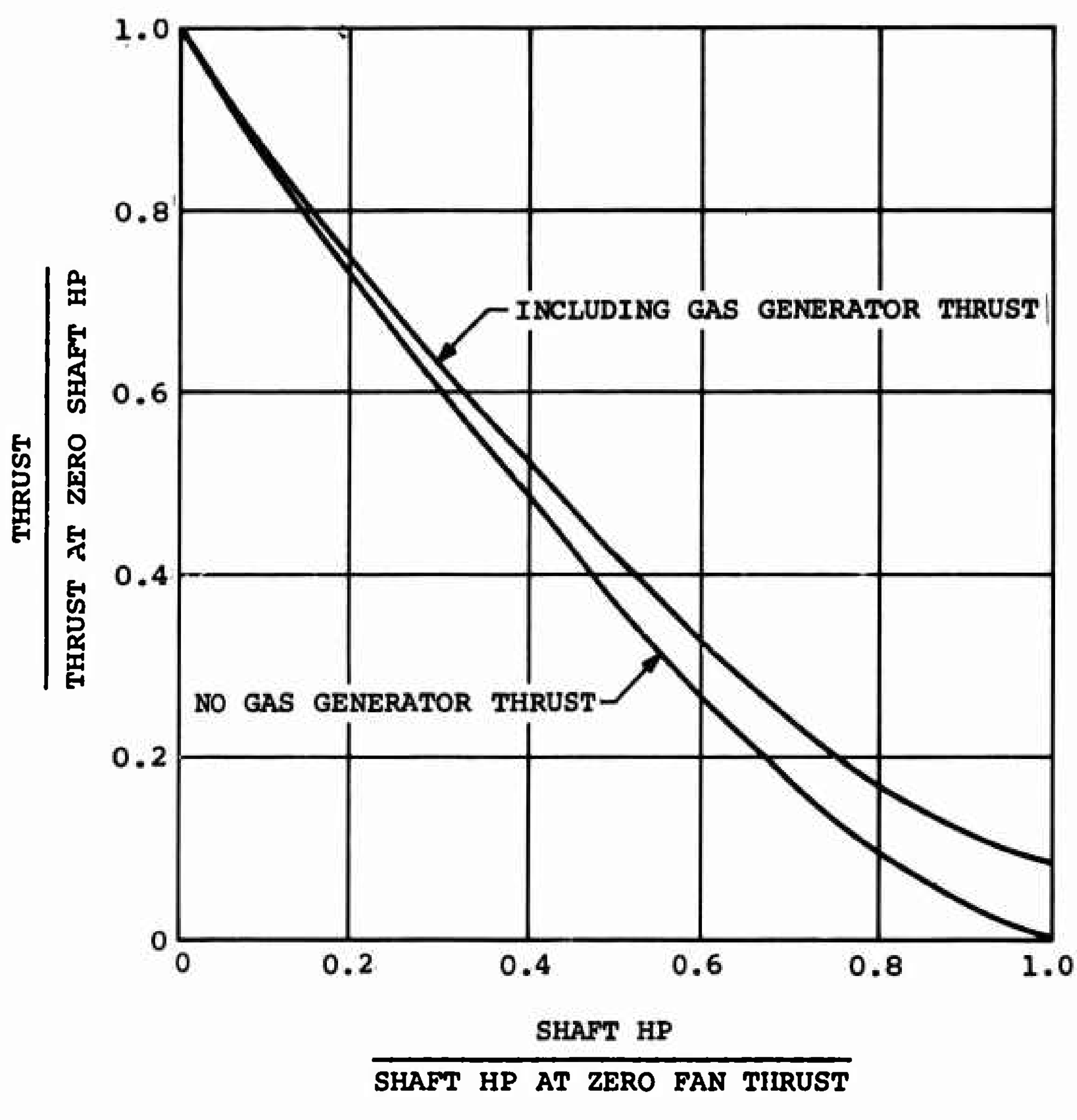


Figure 146. Generalized Thrust-Horsepower Relationship for Fan With Variable Inlet Diaphragm (Inlet Baffles)

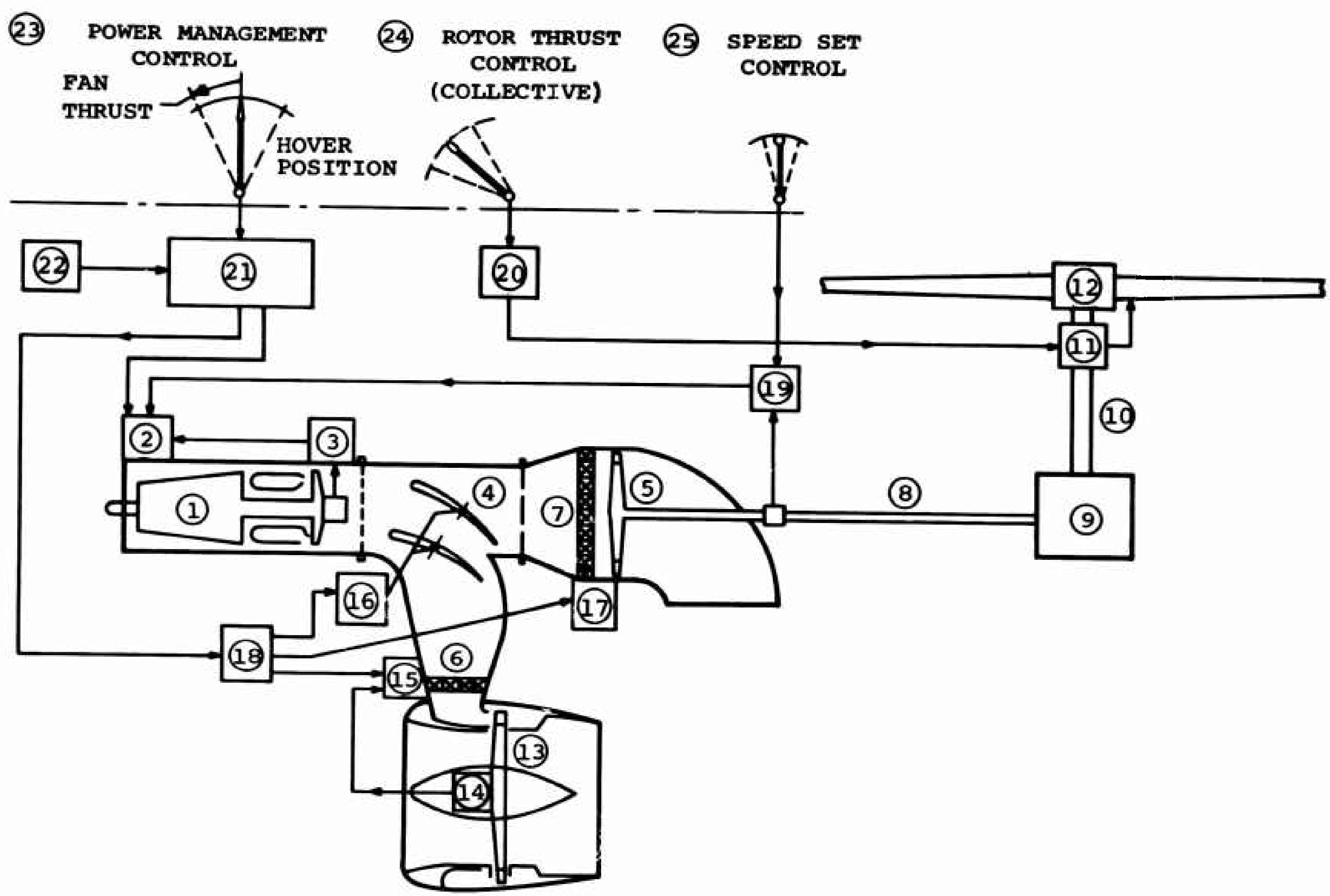


Figure 147. Basic Propulsion System Control Schematic for Integrated Gas-Coupled Systems 1a and 1b

Description of Basic Control System for Integrated Gas-Coupled Systems (Figure 147)

Rotor system speed governor (19) controls rotor speed by means of pilot speed set control (25) together with actual speed input from turbine shaft (8). Governor output regulates gas generator fuel control (2). Gas generator (1) maximum speed is held by topping governor (3) with output to fuel control (2). Cruise fan maximum speed is held by topping governor (14) with output to fan variable admission arch actuator (15).

Thrust of rotor system (12) is varied by pilot's rotor thrust (collective) control (24), as trimmed by action of unit (20) with output to rotor pitch change mechanism (11). Rotor is driven by power turbine (5) through reduction gearing (9) and rotor shaft (10).

Thrust of fan system (13) and division of gas generator power between rotor and fan is varied by pilot's power management control (23) through power management scheduler (21) and power management control unit (18). Dynamic pressure sensor unit (22) provides flight speed input to scheduler (21). Scheduler (21) provides a modulating input on rotor thrust control to trimmer unit (20). Power management control unit (18) coordinates relative settings of diverter valve vanes (4) and variable mission arch systems (6) and (7) through actuators (16), (15), and (17). Diverter valve (4) has three positions: power turbine side closed/fan side open, power turbine side open/fan side closed, and both sides open. Variable arch system (6) regulates available fan gas power, and variable arch system (7) regulates available gas power at the turbine.

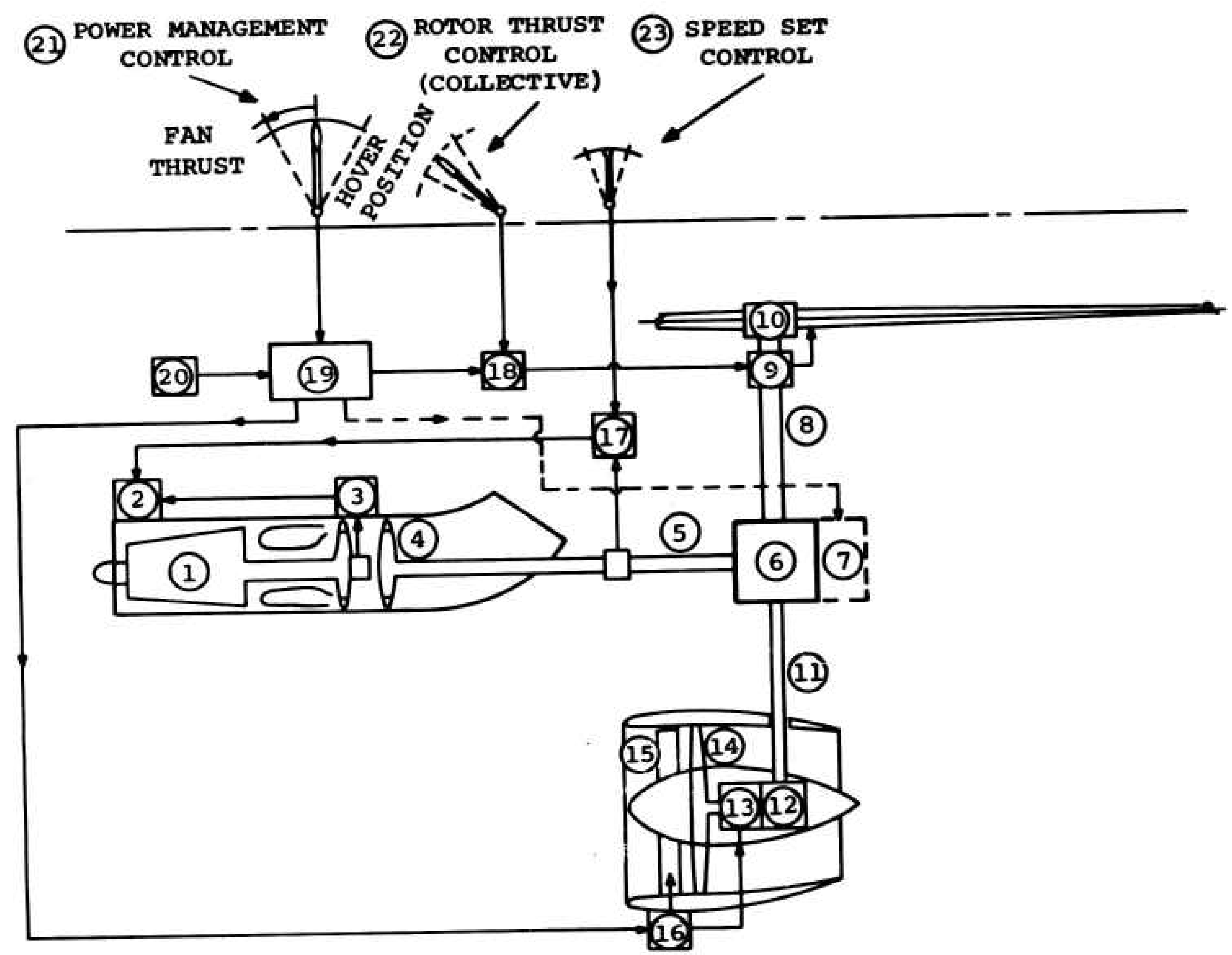


Figure 148. Basic Propulsion System Control Schematic for Integrated Shaft-Coupled Systems 2a and 2b

Description of Basic Control System for Integrated Shaft-Coupled Systems (Figure 148)

System speed governor 17 controls rotor and fan speed by means of pilot's speed set control 23 together with actual system speed input from power turbine output shaft. Governor output regulates gas generator fuel control 2 . Gas generator 1 maximum speed is held by topping governor 3 with output to fuel control 2 .

Thrust of the rotor system 10 is varied by pilot's rotor thrust (collective) control 22 , as trimmed by action of unit 18 , with output to rotor pitch change mechanism 9 . Rotor is driven by power turbine 4 , output shaft 5 , reduction gearing 6 , and rotor shaft 8 . Rotor system 10 can be disengaged by action of decoupling clutch 7 in composite aircraft.

Thrust of fan system 14 is controlled by pilot's power management control 21 through the power management scheduler 19 and pitch change mechanisms 16 for fan variable inlet guide vanes 15 . Fan 14 can be disengaged by the action of decoupling clutch 13 . The fan is powered through fan drive shaft 11 and fan gearbox 12 . Dynamic pressure sensor unit 20 provides flight speed input to scheduler 19 . Scheduler 19 provides a modulating input or rotor thrust control to trimmer unit 18 .

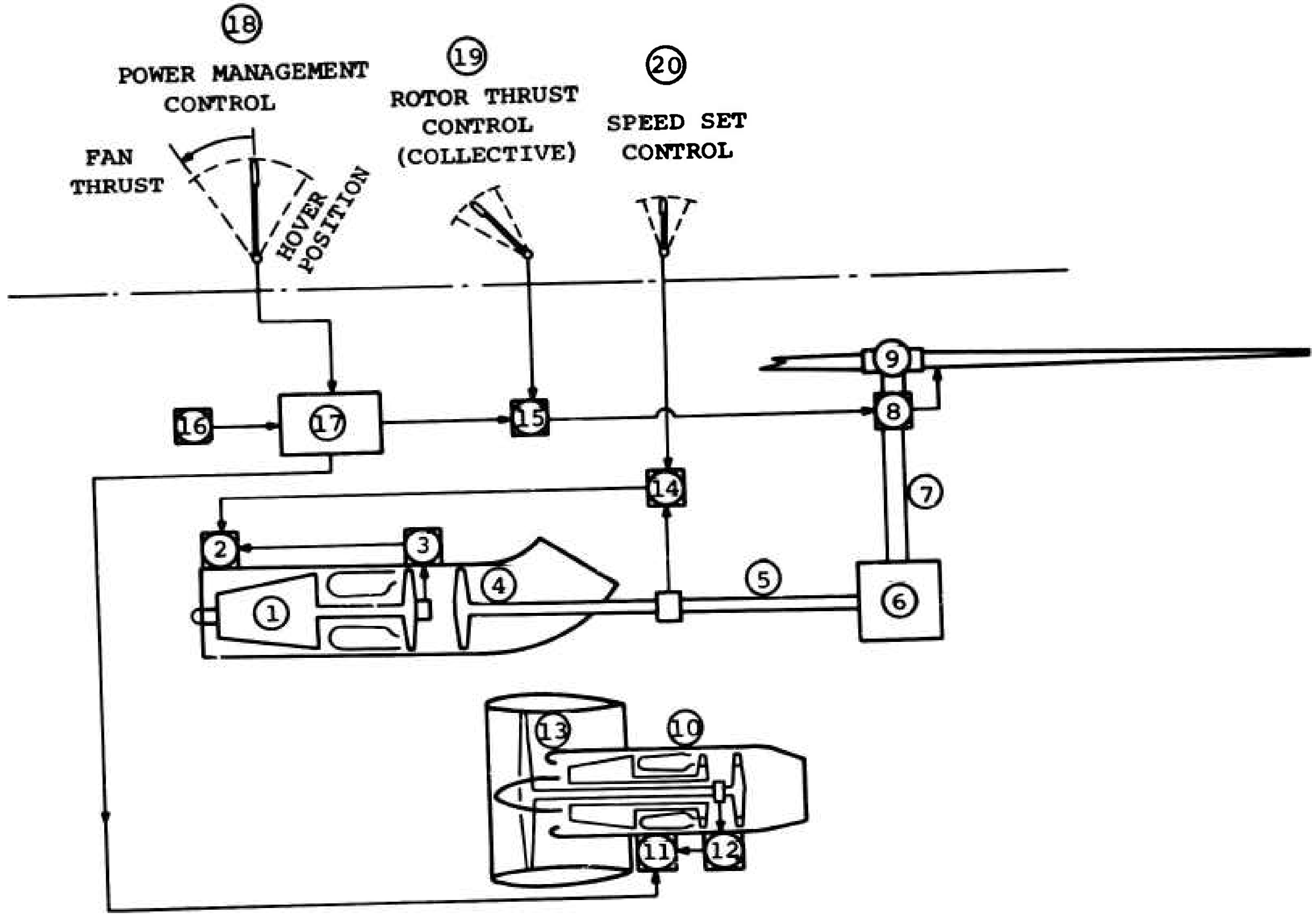


Figure 149. Basic Propulsion System Control Schematic for Independent System 3

Description of Basic Control System for Independent System (Figure 149)

Rotor system speed governor (14) controls rotor speed by means of pilot's speed set control (20) together with actual speed input from turbine shaft (5). Governor output regulates lift system engine fuel control (2). Lift system engine maximum speed is held by topping governor (3) with output to fuel control (2). Maximum speed of cruise fan (13) is held by topping governor (12) acting on fuel control (11) of cruise fan engine (10).

Thrust of rotor system (9) is varied by pilot's rotor thrust (collective) control (19) as trimmed by action of unit (15) with output to rotor pitch change mechanism (8). Rotor is driven by power turbine (4) of lift system engine (1) through reduction gearing (6) and rotor shaft (7).

Thrust of cruise fan system (10) and (13) is varied through action of pilot's power management control (18) through power management schedule (17) on cruise fan engine fuel control (11). Dynamic pressure sensor unit (16) provides flight speed input to scheduler (17). Scheduler (17) provides a modulating input on rotor thrust control to trimmer unit (15).

COMPARATIVE ANALYSIS

COMPARISONS OF OPTIMUM AIRCRAFT CHARACTERISTICS AND SPEED SENSITIVITY STUDY RESULTS

This section presents comparisons, in terms of the five cruise fan propulsion systems studied, of the major characteristics of the 30 optimum aircraft. Summary data from the basic study is shown for comparison in both tabular and graphic form. In addition, two speed-sensitivity investigations were performed as supplements to the basic study so that better comparisons could be made between integrated and independent types of propulsion systems. A discussion of these studies, their results and graphic presentation of the results is given so comparisons of all study results, basic and supplemental, can be made.

Tables XXIX and XXX are summaries of major characteristics for single and tandem rotor optimum aircraft, respectively. The characteristics of the optimum aircraft obtained during the basic study and the results of the speed sensitivity work derived in the supplemental study are presented.

Comparison of the information shown in the tables is presented in Figures 150 through 155, with each figure giving data for a specific type and configuration of aircraft. In these figures the values of cruise speed of the aircraft can also be used as indices of respective productivities, PV (payload x speed), since the payload is a constant 6000-pound value.

Supplemental Study - Speed Sensitivity Study of Integrated Propulsion Systems in Lift/Propulsion-Unloaded and Composite Aircraft

The optimum aircraft of the basic study employing integrated propulsion systems have maximum speeds limited by the use of the available power installed to meet a 6000-foot, 95°F hovering requirement at design gross weight. As a result these aircraft, particularly those of the composite type, are power-limited in high-speed flight and do not necessarily reach their maximum speed potential as an aircraft concept.

The optimum aircraft of the basic study using independent propulsion system 3 have cruise power installed to achieve the maximum speed limits considered appropriate to the concept. These maximum sea level speeds were defined as 270 knots for the compound aircraft, based on rotor aerodynamic limits, and

TABLE XXIX
COMPARISON OF MAJOR CHARACTERISTICS
OF OPTIMUM AIRCRAFT -
TANDEM ROTOR

		Basic Study				Supplemental Study	Basic Study	Supplemental Study
		Integrated				Integrated with Max Speed of Independent	Independent	
		1a	1b	2a	2b	2b	3	3
Propulsion-Unloaded Compound Aircraft	Relative productivity*	82.2	83.6	81.3	78.5	-	54.2	76.6
	Design gross wt (lb)	24,250	24,000	23,700	24,600	-	40,500	26,500
	Empty weight (lb)	15,030	14,790	14,650	15,150	-	29,030	16,225
	Cruise speed, SL (KN)	206	206	199	198	-	262	207
	Disc loading (psf)	11	11	8	8	-	8	8
	Fan bypass ratio	12	9	9	12	-	9	9
	Propulsion wt (lb)	3,155	2,987	3,157	3,447	-	8,370	4,026
	Mission fuel wt (lb)**	2,470	2,460	2,300	2,700	-	4,720	2,625
Lift/Propulsion-Unloaded Compound Aircraft	Relative productivity*	75.9	74.8	64.5	73.6	85.4	76.0	-
	Design gross wt (lb)	26,690	27,330	28,500	27,500	28,680	31,280	-
	Empty weight (lb)	17,298	17,910	18,700	17,850	18,474	20,395	-
	Cruise speed, SL (KN)	219	223	201	219	263	259	-
	Disc loading (psf)	11	11	11	11	11	8	-
	Fan bypass ratio	12	12	12	9	9	3	-
	Propulsion wt (lb)	3,659	4,079	4,230	4,029	4,694	5,592	-
	Mission fuel wt (lb)**	2,642	2,670	3,050	2,900	3,456	4,135	-
Composite Aircraft	Relative productivity*	72.1	69.4	67.6	74.0	78.6	74.8	-
	Design gross wt (lb)	33,400	33,800	35,120	34,710	36,340	38,120	-
	Empty weight (lb)	23,655	24,150	25,310	24,623	25,960	27,375	-
	Cruise speed, SL (KN)	284	279	285	304	340	341	-
	Disc loading (psf)	11	11	11	11	11	8	-
	Fan bypass ratio	12	9	12	6	6	3	-
	Propulsion wt (lb)	4,914	5,323	5,812	5,417	6,117	6,992	-
	Mission fuel wt (lb)**	2,995	2,900	3,060	3,337	3,630	3,995	-

* Relative Productivity = $\dfrac{\text{Design 6000-lb payload} \times \text{cruise speed (KN)}}{\text{Empty weight (lb)}}$

** 100 N Mi Radius at Sea Level

TABLE XXX
COMPARISON OF MAJOR CHARACTERISTICS
OF OPTIMUM AIRCRAFT - SINGLE ROTOR

		Basic Study				Basic Study
		Integrated				Independent
		1a	1b	2a	2b	3
Propulsion-Unloaded Compound Aircraft	Relative productivity*	80.7	82.1	80.9	78.8	58.5
	Design gross wt (lb)	25,000	24,750	24,850	25,550	39,200
	Empty weight (lb)	15,725	15,500	15,720	16,150	27,270
	Cruise speed, SL (KN)	212	212	212	212	266
	Disc loading (psf)	8	8	8	8	8
	Fan bypass ratio	12	12	12	12	6
	Propulsion wt (lb)	3,487	3,389	3,607	3,856	7,771
	Mission fuel wt (lb)**	2,525	2,500	2,380	2,650	5,180
Lift/Propulsion-Unloaded Compound Aircraft	Relative productivity*	79.1	77.6	68.1	77.0	78.9
	Design gross weight (lb)	27,400	27,800	29,030	28,420	30,520
	Empty weight (lb)	17,755	18,338	19,030	18,410	19,732
	Cruise speed, SL (KN)	234	237	216	237	260
	Disc loading (psf)	11	11	11	11	8
	Fan bypass ratio	12	12	12	9	3
	Propulsion wt (lb)	4,257	4,773	4,796	4,677	5,619
	Mission fuel wt (lb)**	2,895	2,712	3,250	3,260	4,038
Composite Aircraft	Relative productivity*	76.0	72.7	71.5	77.6	81.4
	Design gross weight (lb)	32,350	33,700	33,700	33,650	35,850
	Empty weight (lb)	22,660	23,862	23,750	23,580	25,168
	Cruise speed, SL (KN)	287	290	283	305	341
	Disc loading (psf)	11	11	11	11	8
	Fan bypass ratio	12	9	12	6	3
	Propulsion wt (lb)	5,054	5,754	5,650	5,597	6,808
	Mission fuel wt (lb)**	2,940	3,088	3,200	3,320	3,932

$$* \quad \text{Relative Productivity} = \frac{\text{Design 6000-lb payload} \times \text{cruise speed (KN)}}{\text{Empty weight (lb)}}$$

** 100 N Mi Radius at Sea Level

350 knots for the composite aircraft, based on airframe dynamic pressure limits and critical Mach number.

In the cases of lift/propulsion-unloaded compound and composite aircraft, those having independent propulsion systems generally proved to have the highest relative productivity values in the design mission because their increased cruise speeds with respect to aircraft with integrated systems counted for more in the productivity criteria than their increased empty weights.

To estimate the effects on productivity and vehicle weight between aircraft with integrated and independent propulsion systems having the same maximum speed capability, a speed sensitivity study was performed for two aircraft types having integrated systems. The optimum tandem rotor lift/propulsion-unloaded and composite aircraft of the basic study, both using convertible cruise fan propulsion (integrated system 2b), were selected for this purpose. The following approach was employed for this study:

1. Fix the maximum speeds of the two new aircraft at independent propulsion system aircraft values (270 knots for the compound and 350 knots for the composite).

2. Employ the disc loadings and bypass ratios of the optimum aircraft in the basic study for both new cases.

3. Hold the hover capability specifically at the 6000-foot, 95°F level. This rule results in a torque-limited rotor drive system, but directly matches the capability of the comparable independent propulsion system aircraft.

4. Assess the required changes in empty weight, fuel for the 100-nautical mile radius design mission with 6000-pound payload, the resulting gross weight, and the relative productivity. Compare the resulting aircraft with those employing independent propulsion systems and with the lower speed integrated system vehicles of the basic study.

Tables XXXI and XXXII show the results of the study.

TABLE XXXI
INTEGRATED/INDEPENDENT PROPULSION SYSTEMS COMPARISON - TANDEM ROTOR LIFT/PROPULSION-UNLOADED COMPOUND AIRCRAFT

Item	Basic Study Integrated (2b)	Integrated (2b)	Independent (3)
Maximum speed, SL (knots)	226	270	270
Disc loading (psf)	11	11	8
Bypass ratio	9	9	3
Airframe wt (lb)	10,886	10,845	11,868
Propulsion wt (lb)	4,029	4,694	5,592
Fixed equipment wt (lb)	2,935	2,935	2,935
Empty wt (lb)	17,850	18,474	20,395
Fixed useful load (lb)	750	750	750
Payload (lb)	6,000	6,000	6,000
Design mission fuel (lb)	2,900	3,456	4,135
Total useful load (lb)	9,650	10,206	10,885
Design gross wt (lb)	27,500	28,680	31,280
Useful load ratio (%)	35.0	35.6	34.9
Rotor diameter (ft)	39.9	40.7	49.6
Rated hover SHP, SL Std static	6,000	6,300	6,400
Rated cruise fan thrust/ aircraft, SL std, static (lb)	9,180	12,230	10,040
Cruise speed, SL (knots)	219	263	258.5
Relative Productivity $\frac{P \times V_{cr}}{EW}$	73.6	85.4	76.0

Supplemental Study - Speed Sensitivity of an Independent Propulsion System in a Propulsion-Unloaded Aircraft

Results of the basic study defining optimum aircraft showed the relative productivity and gross weight characteristics of the optimum single and tandem rotor propulsion-unloaded aircraft with independent propulsion systems to be poor compared to those of other aircraft configurations. These results are attributed to the 270-knot high-speed requirement imposed on the type in the basic study work. The attempt to design these aircraft for such a high speed, on the basis of rotor aerodynamic limits alone, resulted in extremely high empty and gross weights and

TABLE XXXII INTEGRATED/INDEPENDENT PROPULSION SYSTEMS COMPARISON - TANDEM ROTOR COMPOSITE AIRCRAFT			
Item	Basic Study Integrated (2b)	Integrated (2b)	Independent (3)
Maximum speed, SL (knots)	312	350	350
Disc loading (psf)	11	11	8
Bypass ratio	6	6	3
Airframe wt (lb)	16,271	16,908	17,448
Propulsion wt (lb)	5,417	6,117	6,992
Fixed equipment wt (lb)	2,935	2,935	2,935
Empty weight (lb)	24,623	25,960	27,375
Fixed useful load (lb)	750	750	750
Payload (lb)	6,000	6,000	6,000
Design mission fuel (lb)	3,337	3,630	3,995
Total useful load (lb)	10,087	10,380	10,745
Design gross weight (lb)	34,710	36,340	38,120
Useful load ratio (%)	28.9	28.6	28.2
Rotor diameter (ft)	44.9	45.9	54.9
Rotor hover SHP, SL std static	8,180	8,640	7,920
Rated cruise fan thrust/aircraft, SL std static (lb)	11,200	14,000	12,280
Cruise speed, SL (knots)	304	340	341
Relative productivity $\frac{PxV_{cr}}{EW}$	74.0	78.6	74.8

in low relative productivities. High weights partly resulted from large cruise fan thrust requirements and very high rotor area requirements, the latter being due to the rotor lift requirements while turning at slow rpm at the high aircraft speed.

To provide a better comparison of independent and integrated propulsion systems in this aircraft type, a study was conducted of a revised tandem rotor aircraft with an independent fan system sized for a lower maximum speed of 210 knots. This value of high speed was selected to be closely comparable with those of the optimum propulsion-unloaded aircraft incorporating integrated propulsion systems so that relative productivities could be compared.

The following approach was used for the study of the new aircraft:

1. Fix maximum speed at 210 knots at sea level commensurate with other propulsion-unloaded aircraft. Size the cruise fans for this speed.

2. Employ the disc loading (8) and bypass ratio (9) selected for the original 270-knot high-speed aircraft.

3. Hold the hover capability of the lift system at the 6000-foot, 95°F level commensurate with the other aircraft.

4. Evaluate the required gross and empty weights and the relative productivity in the design mission against those of the other tandem rotor types with integrated propulsion systems.

The results of the study are shown in Table XXXIII.

Discussion of Results

Propulsion-Unloaded Aircraft - A comparison of the relative productivity values resulting from combination of the integrated propulsion systems with propulsion-unloaded aircraft shows a maximum difference of 6 percent. The significance of this difference is somewhat reduced by consideration of the cumulative calculation accuracy resulting from the optimization process. The gas-coupled 1b system appears to yield marginal relative productivity advantages (about 1 to 3 percent); however, it can be concluded from the above considerations that the relative productivity of the systems is essentially the same. The gross weights of these aircraft also are remarkably close, the maximum difference being 4 percent in the tandem case and 3 percent in the single rotor case. Empty weights vary about the same amount.

In one case, the convertible propulsion system 2b aircraft, there is a small increase in gross and empty weights for both the tandem and single rotor cases. This is due to a combination of higher propulsion system weights and mission fuel weights compared to those of other integrated systems. With this propulsion system, the increased fuel requirement is attributed to higher specific fuel consumption when operating principally as a shaft turbine engine due to a nozzle design optimized for cruise

TABLE XXXIII

INDEPENDENT/INTEGRATED PROPULSION SYSTEMS COMPARISON OF TANDEM ROTOR PROPULSION-UNLOADED AIRCRAFT AT APPROXIMATELY THE SAME MAXIMUM SPEED

Propulsion System:	Independent 3	Integrated 1a	Integrated 1b	Integrated 2a	Integrated 2b
Maximum speed, SL (Knots)	210	210	211	203	203
Disc loading (psf)	8	11	11	8	8
Bypass ratio	9	12	9	9	12
Airframe weight (lb)	9,264	8,940	8,868	8,558	8,768
Propulsion weight (lb)	4,026	3,155	2,987	3,157	3,447
Fixed equipment wt (lb)	2,935	2,935	2,935	2,935	2,935
Empty wt (lb)	16,225	15,030	14,790	14,650	15,150
Fixed useful load (lb)	750	750	750	750	750
Payload (lb)	6,000	6,000	6,000	6,000	6,000
Mission fuel (lb)	2,625	2,470	2,460	2,300	2,700
Useful load (lb)	9,375	9,220	9,210	9,050	9,450
Design gross wt (lb)	25,600	24,250	24,000	23,700	24,600
Rotor diameter (ft)	45.1	37.6	37.4	43.4	44.2
Rated hover SHP, SL Std, Static	5,100	5,150	5,110	4,640	4,750
Rated cruise fan thrust, SL Std Static (lb)	2,780	2,965	2,645	2,210	7,850
Cruise speed, SL (knots)	207	206	206	199	198
Relative productivity $\frac{P \times V_{cr}}{EW}$	76.6	82.2	83.6	81.3	78.5

flight. Reference to the power required curves shown later in this section indicates that a high percentage of installed power is being delivered to the rotor at the cruise condition because it still supplies all the lift in cruise flight. This situation is unique to the propulsion-unloaded aircraft type.

The somewhat greater propulsion system weight of the system 2b arrangement reflects a physical design constraint of the convertible engine type. The fan size and installed thrust in this case are functions of basic gas generator sizing, with the latter based on the design hover power requirement. In the other integrated propulsion systems it is possible, because of configuration, to tailor fan size directly to cruise thrust requirements.

Fan bypass ratio selections for these propulsion-unloaded aircraft are all at the high end of the study range (9 or 12). The optimum aircraft selection plots for this type, presented previously, indicate the insensitivity of the relative productivity index to bypass ratio selection. The insensitivity in this case is due to the small amount of cruise fan power, relative to rotor power, required at the cruise-flight condition of this aircraft type. The effect of the bypass ratio on weight is therefore small in terms of the total empty weight value applied in the relative productivity evaluation.

Results show that the aircraft with integrated systems have sea level cruise speeds in the approximate range of 200 to 210 knots where gas generators are sized for the 6000-foot, 95°F design hover condition. Very low cruise fan thrust values are required to attain these speeds. In view of these results, the question could be raised generally as to the applicability of integrated propulsion systems for this aircraft type and the tradeoffs involved against use of an advanced type of pure helicopter Rough estimates would indicate that the pure helicopter might achieve a comparable 200-knot cruise speed if equipped with sufficient rotor blade area.

The results of the basic study for propulsion-unloaded aircraft having independent cruise fans are not directly comparable with those of the aircraft discussed above. For integrated system cases maximum speeds are dependent on the power installed for the hover case, but in independent system cases the high speed

was set as an input parameter and sized the cruise fans. Relative productivity is extremely poor in the independent propulsion system cases of the basic study.

An attempt was made to fly these aircraft at a high speed corresponding to estimated aerodynamic rotor limits in an effort to maximize relative productivity. Instead, the very large increase in vehicle empty weight, due mainly to extreme rotor area requirements and very high fan thrust requirements, sharply reduced relative productivity values. In order to place the independent propulsion system on a more comparable basis with integrated systems and to set a more realistic high speed for the aircraft type, a supplemental speed sensitivity study was performed for a sample tandem rotor aircraft. A discussion of the study is given earlier in this section. The maximum speed of the revised aircraft was adjusted to match the values for the integrated system types.

Results show a substantial increase in relative productivity compared to the poor basic study aircraft, but the relative productivity of the new aircraft is still less by about 2 to 8 percent than that of the aircraft with integrated propulsion systems. The lower relative productivity value is due mainly to a higher propulsion system weight, attributable largely to the fact that four gas generators are employed in the independent propulsion system.

Lift/Propulsion-Unloaded Aircraft - With one exception, these aircraft yield very similar mission relative productivity values when using integrated propulsion systems (within 3 percent in the cases of tandem and single rotor aircraft individually.) Highest values are indicated for the tip-driven gas-coupled system 1a. The exception is the remote shaft-coupled propulsion system 2a aircraft where relative productivities are poorer due to slower cruise speeds and generally higher empty weights. This effect is due mainly to the reduced cruise speeds.

In the 2a system the fan propulsive efficiency is less than in the other systems, thus reducing the cruise speed available with a given amount of gas generator airflow. The lower cruise efficiency stems from use of a fixed, open tailpipe sized for the hover condition in this propulsive system. No credit could be taken in this system for cruise propulsive thrust from the shaft turbine tailpipes because of design installation problems that prevented directing the exhaust aft.

The relative productivity values of the 2b convertible propulsion system are 2.5 to 3 percent lower than those of the 1a system because empty weights are slightly higher. The greater part of the weight difference is in the propulsion system. Increased propulsion system weight stems from a 20-percent increase in installed airflow for the 2b arrangement compared to the 1a system.

The increased airflow requirement results, as before, from a hover inefficiency of the 2b system because of sizing the tailpipes for the design cruise condition. The lower bypass ratio selection of 9 for the 2b propulsion system produced the lowest propulsion system weight for this case. The interplay of convertible fan engine weight variation with fan specific thrust and bypass ratio dictated this selection.

It should be noted, however, that in this case the variation of convertible engine weight with bypass ratio for a given thrust requirement is very small. The optimum aircraft charts previously presented show that the relative productivity variation in the higher bypass ratio range is slight, and that for the higher disc loadings there was little to choose between a bypass ratio of 9 and 12.

In the general case, however, selection of cruise fan bypass ratio has a significant effect on relative productivity, because of its influence on the propulsion group part of empty weight. During cruise flight of lift/propulsion-unloaded aircraft, and contrary to the propulsion-unloaded case, most of the power installed goes to the fans. Therefore, variations in major fan parameters exert a significant influence in propulsion and thus on empty weight trends.

For integrated propulsion systems other than the 2b system, large increases in relative productivity with increasing bypass ratios are shown by the trends of the optimum aircraft selection charts. For a particular cruise fan thrust requirement, the total fan system weight varies with bypass ratio selection. A low bypass ratio results in a lighter fan section and a heavier gas generator, and a high-bypass-ratio results in the opposite. This is due to the lower specific thrust characteristics of a low-bypass-ratio system resulting in higher gas generator airflow requirements. In the cases cited above, the high-bypass-ratio selection yielded the lowest total propulsion system weight.

Results of the study of an independent propulsion system aircraft, where cruise fans were sized by the 270-knot maximum speed, show relative productivity values nearly identical to those of the best integrated system aircraft. Higher speeds have a greater influence than the empty weight increases, but the independent system aircraft requires greater empty and gross weights to attain these productivity values.

In the high-speed independent system aircraft the optimum bypass-ratio selection was the lowest of the study range. Here the relatively lower weight of the low-bypass-ratio fan system resulted from the predominating effect of the lower fan section weight. The trend of relative productivity decrease with increasing bypass ratio for this independent system case is shown in Figures 95 and 100.

To determine whether an integrated system aircraft would produce a higher relative productivity value if maximum speed were increased to the 270-knot independent system aircraft value, a supplementary speed sensitivity study of the 2b system was performed.

Results show that for the new aircraft a substantial increase of 16 percent is gained in relative productivity with respect to the lower speed aircraft of the basic study. A moderate increase in empty and gross weights also results. In addition, the higher speed convertible fan aircraft has superior characteristics to the independent propulsion system aircraft in terms of relative productivity (by 12 percent), and of empty and gross weights (5 and 4 1/2 percent, respectively). It is believed that these trends would be typical for the other aircraft employing integrated propulsion systems.

<u>Composite Aircraft</u> - Optimum composite aircraft of the basic study using integrated propulsion systems display a greater variation of relative productivity with propulsive configuration than other aircraft types. The total range of difference is over 9 percent for tandem rotor and more than 8 percent for single rotor aircraft. The aircraft with the 2b convertible cruise fan systems shows the best relative productivity of the integrated types by a small margin in both cases. The wider variations noted above are due to greater differences in both cruise speeds and empty weights for the various integrated propulsion system aircraft. The higher cruise speeds of composite aircraft relative to the other types emphasize differences in weight and performance characteristics of

propulsive systems. Since no rotor power is employed in cruising flight, the fans use all the available power.

The 2b convertible cruise fan aircraft rates higher in productivity because of a cruise speed advantage. The higher speed of this aircraft results from a higher thrust available at the cruise condition based on the selection of a lower bypass-ratio fan. The static thrust available from this fan system is approximately the same as in other integrated systems; however, the lower bypass-ratio fan gives a slower fall-off of thrust with aircraft forward speed.

The optimum aircraft selection charts for this case presented previously show that relative productivity peaks near the lower part of the bypass-ratio range studied, although the variations with bypass ratios are not large. The selection of bypass ratio 6 in this system reflected the lowest propulsion system weight and the lowest convertible engine weight due to the same total effects of fan specific thrust characteristics and bypass ratio on weight noted previously for the lift/propulsion unloaded aircraft.

An additional factor contributing to the higher speed of the 2b convertible fan aircraft relative to the 2a remote fan case was the selection of an engine nozzle configured for the cruise flight condition, thus providing a higher cruise propulsive efficiency in the 2b system.

The 2a remote shaft-coupled propulsion system shows as the poorest of the integrated system cases because of a relatively low propulsive efficiency as noted earlier in discussion of lift/propulsion-unloaded aircraft.

Composite aircraft equipped with independent propulsion systems exhibit relative productivity values slightly greater than those of the basic study aircraft with integrated systems. The higher cruise speeds of the independent system aircraft, based on propulsion system sizing for a 350-knot maximum speed, increased the productivity values at the expense of significantly higher empty and gross weights. This result is generally similar to that of the lift/propulsion-unloaded aircraft cases discussed previously.

The low-bypass-ratio selection for the independent fan aircraft is again due to the same fan and gas generator weight trends

with bypass ratio and specific thrust noted previously for lift/propulsion-unloaded aircraft.

To determine again the effect on relative productivity and aircraft weights of increasing the maximum speed of an integrated system aircraft to match that of the independent system aircraft, a supplemental study was performed. A discussion of this study is given earlier in this section. The results, shown for a tandem rotor aircraft using a 2b convertible system, indicate that the new higher speed 2b system aircraft has a relative productivity 6 percent greater than the comparable lower speed aircraft of the basic study for a gross weight penalty of just under 5 percent. In addition, comparison on the same maximum speed basis shows the convertible fan system aircraft surpasses the independent system vehicle by 5 percent in relative productivity with empty and gross weights about 5 percent less.

These results clearly show, as found in the lift/propulsion-unloaded case discussed earlier, that the integrated system studied is superior to the independent propulsion system on the comparative basis employed herein. It is believed that similar studies for the other integrated propulsion systems would yield generally similar results.

COMPARISON OF OPTIMUM AIRCRAFT POWER REQUIREMENTS

Figures 156 through 161 present comparisons, each covering the five propulsion systems considered, of installed rotor shaft power and cruise fan thrust requirements through the operating speed range of the 30 optimum aircraft of the basic study. The installed rotor powers are referred directly to the rotor hubs and thus represent requirements of an installed propulsion system. The installed thrust values are net thrusts referred directly to the fan hubs; they also represent requirements of an installed propulsion system.

To relate the installed data given in the curves noted above to uninstalled values, correction factors taking into account the efficiencies of the various drive system components of each propulsion system type, and thus referring power requirements back to the point of engine power output, are given herein. The point to which uninstalled power is referred is the same for each propulsion system: the power turbine output shaft. In the case of gas-coupled systems a factor has been included for duct, diverter valve, and variable admission arch systems

upstream of the common power turbine; however, the efficiency of the power turbine itself has not been included so that a fair comparison with shaft-coupled systems, where shaft turbine free power turbine efficiencies are also omitted, can be effected. The 60 horsepower in the correction factors is the constant value employed for aircraft accessory drive power in the study. The following tabulation shows the factors with which the rotor shaft power requirement values of the curves should be modified to obtain uninstalled power values as defined above.

Power Correction Factors on Curve Installed Values

Propulsion System	Uninstalled Power Value (HP)
Gas-Coupled Integrated (1a, 1b)	$1.108\ HP_I + 60$
Shaft-Coupled Integrated (2a, 2b)	$1.064\ HP_I + 60$
Independent (3)	$1.064\ HP_I + 60$

where

HP_I = curve value (installed) (hp)

The fan thrust requirement data shown in the curves can be modified to installed (fan hub) and uninstalled power requirements as follows: Fan installed power for any point is gained through the formula

$$(HP_I)_f = \frac{T_f \times V}{326\eta_f}$$

where

f = fan

T_f = curve value of fan thrust per aircraft (lb)

V = curve value of aircraft flight speed (knots)

η_f = fan efficiency

Fan efficiencies can be taken from the data of Figure 176 for the bypass-ratio values pertaining to the optimum aircraft of interest. The correction factors required to translate an installed fan power requirement to uninstalled values are given in the following tabulation:

Fan Power Correction Factors

Propulsion System	Uninstalled Power Value (HP)
Gas-Coupled Integrated (1a, 1b)	$1.075\ (HP_I)_f + 60$
Shaft-Coupled Integrated (2a)	$1.031\ (HP_I)_f + 60$
Shaft-Coupled Integrated (2b)	$1.016\ (HP_I)_f + 60$
Independent (3)	$1.0\ (HP_I)_f + 60$

where
$(HP_I)_f$ = calculated fan installed power value from above formula

The variations of required power and thrust as shown on these curves are generally similar to the trends shown as samples, and as discussed under power requirements in the configuration studies section of this report. Differences in the requirements of aircraft using the four integrated propulsion systems are in general quite small for a specific aircraft type. This is due to the small differences in the major parameters of weight and disc loading that govern power requirements in the aircraft using these systems. Reference to the summary data for the 30 optimum aircraft in the previous section is an aid in interpreting the power/thrust requirement trends shown in the curves. The independent system 3 aircraft requirements are in most cases substantially greater, since these aircraft are much heavier because of the high-speed requirement imposed. Variations in the hover power requirements are a function of variations in aircraft weight, disc loading selection, drive system losses, and download force from rotor wash on the wings of wing-equipped types.

COMPARISON OF OPTIMUM AIRCRAFT PAYLOAD-DISTANCE CAPABILITY

The payload-range availability of the 30 optimum aircraft is shown in graphical form in Figures 162 through 167. Comparisons of aircraft using the five propulsion systems studied are shown on each curve for a specific aircraft type and configuration.

The design point for all aircraft in accordance with study ground rules was a sea level 6000-pound payload, 100-nautical-mile-radius mission. This is shown in the curves as an equivalent 200-nautical-mile-range mission, and the common design point is indicated on each plot. All the aircraft were designed with a fuel capacity tailored to this requirement, and

thus the sloped load-distance tradeoff lines terminate at the 200-nautical mile point. The plots are completed by the near-vertical maximum fuel cutoff lines.

The slopes of the load-distance trade lines give relative indices of the overall mission specific ranges (nautical miles per pound of fuel) for the various aircraft/propulsion system combinations. Flatter slopes indicate better overall specific range and conversely lesser payload increases per unit decrement in range. The aircraft all have different gross weights, however, and greater load capability for a given distance is generally gained at the expense of a higher aircraft gross weight, as can be seen by comparison of the payload-range data with the optimum aircraft gross weights presented in the figures.

In the cases of all types of aircraft employing the integrated propulsion systems, the payload-range capability is similar within a type.

The maximum spread of difference in zero-range payload is about 7 percent. This results because gross weights and specific range values are quite similar for a given aircraft type.

The aircraft employing independent propulsion systems show, in general, significant differences from those with integrated systems in payload-range capability at distances below the design point value. This results from the higher gross weights and poorer specific range characteristics of the optimum independent system 3 aircraft. Though of secondary importance to weight at very short ranges, the bypass ratio selections for system 3 aircraft tend to be lower than in other propulsion system cases, thus tending to increase specific fuel consumption, decrease propulsive efficiency, and additionally reduce specific range characteristics.

COMPARISONS OF OPTIMUM AIRCRAFT CRUISE TIME VERSUS HOVER TIME CAPABILITY

Figures 168 through 173 present the results of an analysis of comparative availability of time in hover flight versus time in cruising flight for the optimum aircraft. The various gross weights are as defined in the basic study. The results are presented in separate plots for the particular aircraft types and configurations, with the five propulsion systems compared

in each case. A general relative comparison of mission tradeoffs available for the aircraft/propul ion system combinations studied is provided.

All aircraft were sized by the 6000-pound, 100-nautical-mile-radius design mission employing a profile which allowed only the standard fuel provisions for vertical takeoff and landing appropriate to the start, midpoint, and end of the mission. Otherwise, no specific amount of time in hover flight was included in the designing profile. Since in all cases the design point of these aircraft is essentially at the point of zero hover time on the curves, available hover time is a fall-out characteristic in each case.

Cruise time capability in any specific aircraft case of zero hover time is basically the fixed 200-nautical-mile total design point distance divided by the cruise speed of that aircraft. It is thus the aircraft reaction time to accomplish a 200-nautical-mile-range mission. In proceeding along a typical trade line of constant gross weight in the direction of shorter range capability, that is, less cruise time capability, the point of zero distance is reached when all fuel is expended during hover flight. Differences in hover capability between various aircraft at any point are a function of relative differences in total fuel available for hovering and efficiency of fuel use during hover. Hovering efficiency is primarily a function of the disc loading selected during the optimization procedure in the basic study.

In general, the data shows the cruise times are similar where cruise speeds are similar. The consistently lower cruise times of aircraft using independent propulsion systems result from the higher cruise speeds employed. The differences in hover time vary with fuel available and disc loading. Aircraft using independent propulsion system 3 have the greatest hover time capability, in spite of generally incorporating lower disc loading values than the other aircraft, because they have so much more fuel available.

COMPARISONS OF OPTIMUM AIRCRAFT FERRY RANGES

The ferry ranges of the aircraft defined as optimum from the basic study results are fall-out data, since vehicle design points were in all cases fixed by the 6000-pound payload, 100-nautical-mile-radius sea level mission. Ground rules as to takeoff weight limitations and mission profiles for the

ferry case were stipulated in a previous section of this report.

A tabular comparison of the calculated ferry ranges of the 30 optimum aircraft is presented in Tables XXXIV and XXXV for tandem and single rotor design configurations, respectively. This data is presented in each case for the three aircraft types covered against the five propulsion systems considered in the study.

The ferry range figures in general show trends consistent with the variation of major influencing aircraft parameters and the differences of aircraft type. Ranges increase in the progression of propulsion-unloaded, lift/propulsion-unloaded, and composite types. This is mainly because fuel available for the mission increases in the same progression due to successively increasing ferry takeoff gross weights, and for a given propulsion system the mission specific range varies only a small amount between aircraft types.

Generally, where better aircraft lift/drag ratio characteristics are available in a type, the increased average gross weight tends to nullify the potential gains in specific range. Range differences do occur, however, in some cases where takeoff gross weight is limited by something other than a flight limit load factor of 2.0, such as available power, thus placing a restriction on the amount of fuel available.

Since for aircraft of a given type using integrated propulsion systems there are no major differences in takeoff gross weights, ferry fuels or specific ranges, the ferry ranges of vehicles in both single and tandem configurations vary little among themselves. The influence of bypass ratio alone in changing specific range values through changes in propulsive efficiency and specific fuel consumption appears to show to a minor degree in the data presented. Since bypass ratios are consistently on the high side of the study range (usually 9 or 12) on all integrated system aircraft, differences in specific range should be small.

Aircraft with independent propulsion systems show poorer ferry ranges in spite of greater fuel availability. This poor specific range performance is due to relatively greater average aircraft weights and, in most cases, to a lower fan-bypass-ratio selection from the basic study. The lower bypass-ratio selection by itself tends to decrease propulsive efficiency and

TABLE XXXIV

COMPARISON OF OPTIMUM AIRCRAFT FERRY RANGES - TANDEM ROTOR CONFIGURATIONS

Propulsion System	1a	1b	2a	2b	3
Propulsion-unloaded aircraft					
Fan bypass ratio	12	9	9	12	9
Ferry range (N Mi)	1,580	1,600	1,850	1,820	1,770
Ferry takeoff gross weight (lb)	34,200	34,000	35,800	36,900	60,800
Total ferry fuel (lb)	15,900	15,930	18,355	18,927	26,397
Ratio, ferry TO GW/Design GW	1.41	1.42	1.50	1.50	1.50
Lift/propulsion-unloaded aircraft					
Fan bypass ratio	12	12	12	9	3
Ferry range (N Mi)	2,000	1,930	2,080	1,960	1,950
Ferry takeoff gross weight (lb)	40,035	40,955	42,750	41,250	46,920
Total ferry fuel (lb)	20,157	20,551	21,460	20,860	23,805
Ratio ferry TO GW/Design GW	1.50	1.50	1.50	1.50	1.50
Composite aircraft					
Fan bypass ratio	12	9	12	6	3
Ferry range (N Mi)	2,327	2,204	2,363	2,283	2,020
Ferry takeoff gross weight (lb)	50,100	50,800	52,600	52,000	57,100
Total ferry fuel (lb)	23,552	24,625	24,540	24,607	26,795
Ratio, ferry TO GW/Design GW	1.50	1.50	1.50	1.50	1.50

Propulsion System Key:
1a - Integrated gas-coupled, tip-driven
1b - Integrated gas-coupled, hub-driven
2a - Integrated shaft-coupled, remote
2b - Integrated shaft-coupled, convertible
3 - Independent

TABLE XXXV

COMPARISON OF OPTIMUM AIRCRAFT FERRY RANGES - SINGLE ROTOR CONFIGURATIONS

Propulsion System	1a	1b	2a	2b	3
Propulsion-unloaded aircraft					
Fan bypass ratio	12	12	12	12	6
Ferry range (N Mi)	1,990	1,980	1,950	1,925	1,840
Ferry takeoff gross weight (lb)	37,500	37,100	37,300	38,300	58,800
Total ferry fuel (lb)	19,450	19,200	19,170	19,710	28,475
Ratio, ferry TO GW/Design GW	1.50	1.50	1.50	1.50	1.50
Lift/propulsion-unloaded aircraft					
Fan bypass ratio	12	12	12	9	3
Ferry range (N Mi)	2,110	2,030	2,130	2,080	1,940
Ferry takeoff gross weight (lb)	41,100	41,700	43,545	42,630	45,780
Total ferry fuel (lb)	20,805	20,902	21,905	21,640	23,378
Ratio, ferry TO GW/Design GW	1.50	1.50	1.50	1.50	1.50
Composite aircraft					
Fan bypass ratio	12	9	12	6	3
Ferry range (N Mi)	2,348	2,227	2,383	2,305	2,041
Ferry takeoff gross weight (lb)	48,500	50,500	50,500	49,600	53,800
Total ferry fuel (lb)	23,210	23,908	24,020	23,665	25,832
Ratio, ferry TO GW/Design GW	1.50	1.50	1.50	1.47	1.50

Propulsion System Key:
- 1a - Integrated gas-coupled, tip-driven
- 1b - Integrated gas-coupled, hub-driven
- 2a - Integrated shaft-coupled, remote
- 2b - Integrated shaft-coupled, convertible
- 3 - Independent

to increase specific fuel consumption characteristics, thereby decreasing the specific range characteristic.

The relatively lighter weight of the lower bypass-ratio cruise fans in aircraft with independent propulsion systems has very little effect on a long-range mission such as the ferry case. Another factor tending to increase independent system fuel specifics and to decrease aircraft specific range is the somewhat lower part-power operation of these units in the ferry mission compared to the operation of integrated systems. This results from the relatively higher values of installed thrust for the independent system to meet the high speeds specified for this type in the basic study.

COMPARISONS OF PROPULSION SYSTEMS

Weights

The five selected propulsion systems installed in the various types of aircraft were compared to determine what major differences existed which would influence the selection of one over another. It should be noted, however, that the systems compared included features which were a result of the study ground rules, and the resulting differences should be viewed with these rules in mind.

The propulsion system weights were divided into two major categories: power generation and power transmission. Figures 174 and 175 show these weights for each of the propulsion systems for the tandem and single rotor aircraft respectively. These figures show, for the integrated systems sized only for hover, small differences in the propulsion system power generation weights. Slightly larger differences occur in power transmission weights resulting in a similar larger difference in total weights.

If only total propulsion system weight and hover requirements design conditions are considered, the tip-turbine gas-driven system was lighter than the other integrated systems. This was partially due to the design decisions locating the components of this system so that only very short hot ducts were required. It should be noted, however, that other design criteria, such as maintainability and vulnerability, could be more important in the selection of a propulsion system than the small difference in weight.

Since the independent propulsion systems were sized for cruise rather than hover requirements, the data shown for these systems cannot be directly compared to that of the integrated systems. A separate sensitivity study was made to determine the effects of sizing the convertible cruise fan integrated system 2b for cruise and was presented in a previous section of the comparative analysis. The propulsion system weights resulting from this sensitivity study are shown in Figure 174.

Sizing the 2b propulsion system for 270 and 350 knots cruise speeds increased the weights 14 percent for the lift/propulsion-unloaded aircraft and 17-1/2 percent for the composite aircraft. However, these weights were 24 and 35 percent less than the corresponding independent system 3 weights. These differences were significant and, as with the earlier productivity comparisons resulting from this sensitivity study of these systems, represent a sizable gain.

The effect of a maximum cruise speed of 270 knots as a ground rule in designing the propulsion-unloaded aircraft using the independent propulsion system resulted in a large power required (brute force) and in an unusually large rotor area. Thus the power generation weight and the total propulsion weight of this design were much higher than those of the other systems (Figures 174 and 175). A sensitivity study was performed at a maximum speed comparable to those of the integrated systems to present a more effective comparison of propulsion systems; it was presented in a previous section of the comparative analysis. The total propulsion system weight of this design with a maximum speed of 210 knots is shown in Figure 174. This reduction of forward speed requirement of 22 percent resulted in a reduction of propulsion system weight of 52 percent.

Propulsion Efficiency

Figure 176 gives the variation of propulsive efficiency for the cruise fans used in the optimum aircraft versus forward speed. For the purposes of this study the following definition of fan propulsive efficiency was used:

$$\eta_p = \frac{1116 \times M \times F_n}{550 \times SHP}$$

where

η_p = fan propulsive efficiency

M = flight Mach number

F_n = installed net fan + fan turbine thrust (lb)

SHP = input horsepower at the fan hub (hp)

It should be pointed out that the thrust value used in the above expression is net thrust and includes both fan thrust and power turbine residual exhaust thrust as well as fan nacelle drag.

Propulsive efficiencies as defined above for all but the remote shaft-coupled fan system 2a vary from about 50 percent at 150 knots to almost 70 percent at 400 knots. The variation between systems, again with the exception of the 2a system, is quite small (about 8 percent at 150 knots and only 3 percent at 400 knots). The deviation of system 2a from this pattern is due to the design of the turbine exhaust. Because of the physical location of this exhaust, no use could be made of residual thrust; therefore, the nozzle is a complete expansion nozzle giving maximum efficiency in hover flight, but producing a non-optimum match (lower propulsion efficiency) in cruise flight.

Installation and Maintenance

In the design integration section, problems of hot gas duct installation were discussed and propulsion system locations were chosen to minimize these problems. The large size and high temperature of the hot gas ducting remain as a disadvantage in that slip joints, ball joints, and bellows are a potential source of leakage requiring constant inspection. In addition, large structural installation access panels would have to be provided, thereby increasing weight and reducing structural efficiency. Adequate shrouding must be provided to protect the adjacent structure and to maintain the normal area temperature below 250°F through forced circulation.

Shaft-driven systems will require precision dynamic balancing to eliminate transmission of vibration to vital components so as not to adversely affect the system reliability and increase maintenance requirements. However, when properly balanced,

shaft systems should add no new installation or maintenance problems. On a comparative basis the convertible shaft system with its inherent compactness and external location would reduce installation problems and access requirements, thus providing improved maintainability. The entire propulsion system can be contained in two interchangeable pods requiring only an interconnecting shaft to the rotor-drive system, and would be qualified in this configuration as such.

The independent system has the advantage of not requiring a new type of power management system (variable inlet guide vanes, clutches, variable turbine arches, etc.); but because the number of systems is doubled, additional maintenance will be required and a reduced level of reliability may result.

Foreign Object Damage

It is difficult to compare the basic propulsion systems on the basis of foreign object damage (FOD), since this factor is dependent on aircraft installation as well as on engine design. Some differences may be predicted, however, on the basis of number, location, and the type of engine inlet.

For gas coupled systems 1a and 1b, normal inlets are associated with the integrally mounted gas generators, and no unusual problems are created by the remote fans. However, shaft-coupled systems 2a and 2b may be more susceptible to foreign object damage. Large objects, such as paper or birds, may impact on the nonrotating variable inlet guide vanes and block the fan or engine inlet. These same objects hitting a rotating fan blade might be shredded and pass through the engine without loss of power or damage to the engine.

It would appear that the independent system 3 would also have an increased susceptibility to foreign object damage, due mainly to the doubling of the number of propulsion systems. Thi tendency would be magnified in the case of augmented vertical lift through use of a vectored thrust system due to increased downwash velocities and maximum power conditions on all four systems during takeoff and landing.

Vulnerability

The evaluation of aircraft vulnerability is a complicated process and requires such inputs as damage vulnerable area, aircraft speed and altitude, aircraft armor and structural

shielding, aspect angle, vulnerable component separation, weapon type, etc. Obviously, it is beyond the scope of this study to carry out such an analysis on each of the optimum aircraft. A preliminary assessment of vulnerability was made, however, on the basis of the damage vulnerable area of the propulsion and drive systems only.

The kill probability given in terms of damage vulnerable area is given by:

$$P_k = 1-e^{-A_v n P_h}$$

where

P_k = kill probability

A_v = damage vulnerable area

n = number of rounds

P_h = hit probability

A plot of this equation is shown in Figure 177. Table XXXVI gives the damage vulnerable area of each of the propulsion systems in this study as installed in the tandem and single rotor [illegible]/propulsion-unloaded aircraft. Figure 177 gives an estimate of the vulnerability of these systems.

In comparing these systems, the vulnerability of a shaft versus that of a duct has been treated on a one-to-one basis; however, this may not necessarily be the case. A hit on a shaft or gearbox may not provide a kill; it involves the analysis of what actually constitutes a vulnerable spot. On the other hand, any hit in the duct area releasing the 1500°F hot gas is a potential hazard to the aircraft. This would indicate the use of some sort of weighting system to better assess the vulnerability of the system.

The use of a value of 0.02 for the probability of hit, P_h, is an estimate of an average for this factor. Obviously, the probability of hit varies with altitude, aspect angle, weapon type, speed, etc., and requires computer integration for more accurate analysis. Based on these simplifying assumptions, however, the convertible system seems to be the least vulnerable due to its compactness. Care should be taken in generalizing these results, however, since they are a function of the

TABLE XXXVI DAMAGE VULNERABLE AREA - LIFT/PROPULSION-UNLOADED AIRCRAFT		
Rotor Type	Propulsion System	A_V (Sq Ft)
Tandem Rotor	1a	8.36
	1b	9.31
	2a	8.28
	2b	5.95
	3	7.37
Single Rotor	1a	8.45
	1b	9.40
	2a	8.90
	2b	6.40
	3	7.56

optimum aircraft characteristics. For example, the similarity of the remote fan (2a) and independent (3) systems should make them equally vulnerable. In this case the 2a seems more vulnerable than the 3 system. This is primarily due to the bypass ratio difference (12 for the remote fan and 3 for the independent system) which results in a different exposed area. However, it is generally concluded that the shaft-driven systems are less vulnerable than the gas-driven systems, with the convertible cruise fan being the least vulnerable.

Summary

A summary of relative merits of the five configurations is shown in Table XXXVII; maintenance, reliability, foreign object damage and combat vulnerability are considered; each rating factor has an ideal or maximum value of 25 points. A maximum rating value total of 100 points is thus available. Based on the above cursory analysis, it appears that the convertible fan engine would show the highest rating.

TABLE XXXVII SUMMARY OF RELATIVE MERITS OF THE FIVE CONFIGURATIONS					
Propulsion System Configuration	Maintenance	Reliability	FOD	Combat Vulnerability	Total
la, lb	10	10	20	5	45
2a	15	15	15	10	55
2b	20	20	15	20	75
3	10	10	10	15	45

Availability

Since the propulsion data presented in this report is 1970 state of the art, some discussion as to the probability that these systems can actually be available in that time is in order. It is believed that the performance data for the propulsion systems used in this study is attainable and that the turbine inlet temperature, pressure ratio, and component efficiencies assumed were a conservative estimate of evolutionary change. On the other hand, propulsion system weights were somewhat lower than those predicted by some engine manufacturers, but they seem to be attainable through new materials and designs. Some weight advances have already been shown to be possible through current weight improvement demonstrations. Of particular interest are the power management systems. A number of engine manufacturer preliminary design efforts are currently under way, and in some cases test hardware is being developed.

The use of variable admission arches for turbine power control should present no insurmountable problems. Even though more complex, it is an advance over current partial admission turbine state-of-the-art knowledge. Variable inlet guide vanes for

varying fan thrust will require further test and development to determine the feasibility of their use throughout a range of thrust from zero to maximum.

Since the convertible cruise fan system presented in this report is, in general, an attractive system, it is felt that new problems associated with this design should be researched to make this engine available within the 1970 time frame. Such problems as variable inlet guide vanes and fan declutching are in this category. In general, all systems can be made available with normal evolutionary engine advances.

COMPARISONS OF TILTING AND NONTILTING CRUISE FANS - INDEPENDENT PROPULSION SYSTEM 3

Based on the previous review of the potential design application of vectored thrust fans to the various aircraft types, an example of such application was selected, and an analysis was made to compare the resulting aircraft with one employing nontilting fans. A tandem rotor composite aircraft was selected for this purpose, and the optimum aircraft of this type in the study was employed as a base line for comparison. In this case the fans are mounted on pylons beneath the wing of the aircraft, and they tilt with the wing (see Figure 178). The fans are located so they clear the forward rotor droop envelope in the tilted (hover) position. This places the engine tailpipes very close to the ground; however, there is sufficient clearance for all normal attitudes of aircraft landing. The fans are mounted as close inboard as clearance of the exhaust core from airframe components will permit. It is assumed that no intershafting is employed and that a fast-acting control system caters to a case of sudden fan engine failure in hover flight. Since in hover flight the fan thrust line of action is offset forward from the aircraft center of gravity, it is assumed that the aircraft can be trimmed longitudinally by the rotor control system. The following approach was taken in the comparison:

1. Hold the tilt-fan-aircraft design gross weight, disc loading, cruise fan bypass ratio, and fan size at the values of the optimum nontilt aircraft.

2. Assess the changes in weights of airframe, propulsion, and fuel items resulting from a change to tilting fans.

3. Compute the change in design mission payload and mission relative productivity from the base-line aircraft.

With fans rated for the desired high-speed condition of 350 knots at sea level, about 25 percent of the aircraft weight is supported at the hover condition by fan thrust. Holding a rotor disc loading of 8 psf allows a reduction in rotor diameter of 13.7 percent and a chord reduction of about 12.5 percent. The required airflow and power rating of the shaft turbines is reduced by 21 percent in this case. Table XXXVIII shows the comparison in terms of weight differences. There are no first-order changes in aircraft drag characteristics. Since the rating of the fans is held constant, the cruise and high-speed performance of the aircraft remains essentially the same.

The rotor weight of the tilt-fan aircraft was reduced due to a smaller size and design power requirement. The weight of the tilting wing does not perceptibly change since major characteristics remain constant, and the essential difference is the mounting of the fans inboard beneath the wing panels in a position where inertial or aerodynamic loading on the wings should not have a large effect.

Although the change in rotor diameter would have some influence on fuselage stowage bay and rotor fairing design, the basic size and wetted area of the fuselage does not change, and any weight differences of a detailed nature are beyond the scope of the current analysis. Since the gross weight of the aircraft is held constant, no landing gear weight differences are envisioned. Flight control weight changes occur due to differences in rotor controls with the rotor size change, and the engine section weight varies because of differences in the mounting location of fans in the two cases considered.

The weight of the shaft turbine installation and the transmission drive system to the rotors is reduced since the design powers are lower in the tilt-fan-aircraft case, whereas the weight of the fan installations differ only because of a change in location on the aircraft. Fixed-equipment weight is held at the same ground-rule value employed on all aircraft in the study.

The empty weight of the tilt-fan aircraft is 2745 pounds less than that of the basic aircraft without fan tilt. However, the fuel required for the design mission is approximately

200 pounds more in the tilt fan case. It was estimated that a total of 4 minutes of normal power use of fans during vertical operation in the design mission consumed approximately 310 pounds of fuel, whereas operation at idle power in the non-tilting fan case required about 60 pounds. The reduction of installed lifting power required of the shaft turbines in the tilt-fan case was approximated as equivalent to 50 pounds of fuel for the 4 minutes of operation. The net fuel increment in the tilt-fan case would thus be 310-60-50 or 200 pounds. It should be noted, however, that if the mission profile were changed to require significant time in the vertical flight mode, rather than the 4 minutes of the design mission employed herein, the increase in fuel requirement for the tilt-fan aircraft could drastically alter the results.

Since in the tilt-fan case the empty weight decrease plus net fuel increase amounts to 2545 pounds, the payload that may be carried by the tilt-fan aircraft in the design mission profile is 2545 pounds greater than the basic 6000-pound payload case, or 8545 pounds at the same design gross weight. In terms of the basic criteria previously employed for measurement, relative productivity, or the product of payload and cruising speed per pound of empty weight in the design mission profile, the aircraft employing tilting fans shows a 58-percent improvement over the basic nontilting fan case.

It is expected that generally similar results would be obtained, assuming very low hover times, for other aircraft types where installation of a fan thrust vectoring system appears feasible from a design installation standpoint. It must be decided, therefore, whether the improvements in payload capability or productivity are worth the cost in terms of potential forward area operational problems noted previously.

TABLE XXXVIII

CHARACTERISTICS AND WEIGHT COMPARISON OF AIRCRAFT WITH TILTING AND NONTILTING INDEPENDENT FANS

Item	Tilting Fans	Nontilting Fans
Aircraft type	Tandem rotor composite	Tandem rotor composite
Propulsion system	Independent	Independent
Rotor disc loading (psf)	8	8
Cruise fan bypass ratio	3	3
Design gross weight (lb)	38,120	38,120
Weight supported by rotors, hover, 6000 ft, 95°F (lb)	28,550	38,120
Installed shaft turbine power, SHP, SL std	6,250	7,920
Rotor diameter (ft)	47.4	54.9
Wing area (ft^2)	476	476
Installed fan thrust, SL std(lb)	12,280	12,280
Group weights (lb):		
Rotor	3,529	4,940
Wing	2,062	2,062
Tail	446	446
Body	5,918	5,919
Alighting gear	1,448	1,448
Flight controls	1,983	2,441
Engine section	408	192
Propulsion install. (rotor drive)	1,407	1,607
Drive system (rotor)	2,854	3,745
Fan installation	1,640	1,640
Total fixed equipment	2,935	2,935
Weight empty	24,630	27,375
Fixed useful load	750	750
Fuel (design mission)	4,195	3,995
Available mission payload	8,545	6,000
Design gross weight	38,120	38,120
Relative productivity $\frac{\text{Payload x VCruise}}{\text{Empty weight}}$	118.0	74.8

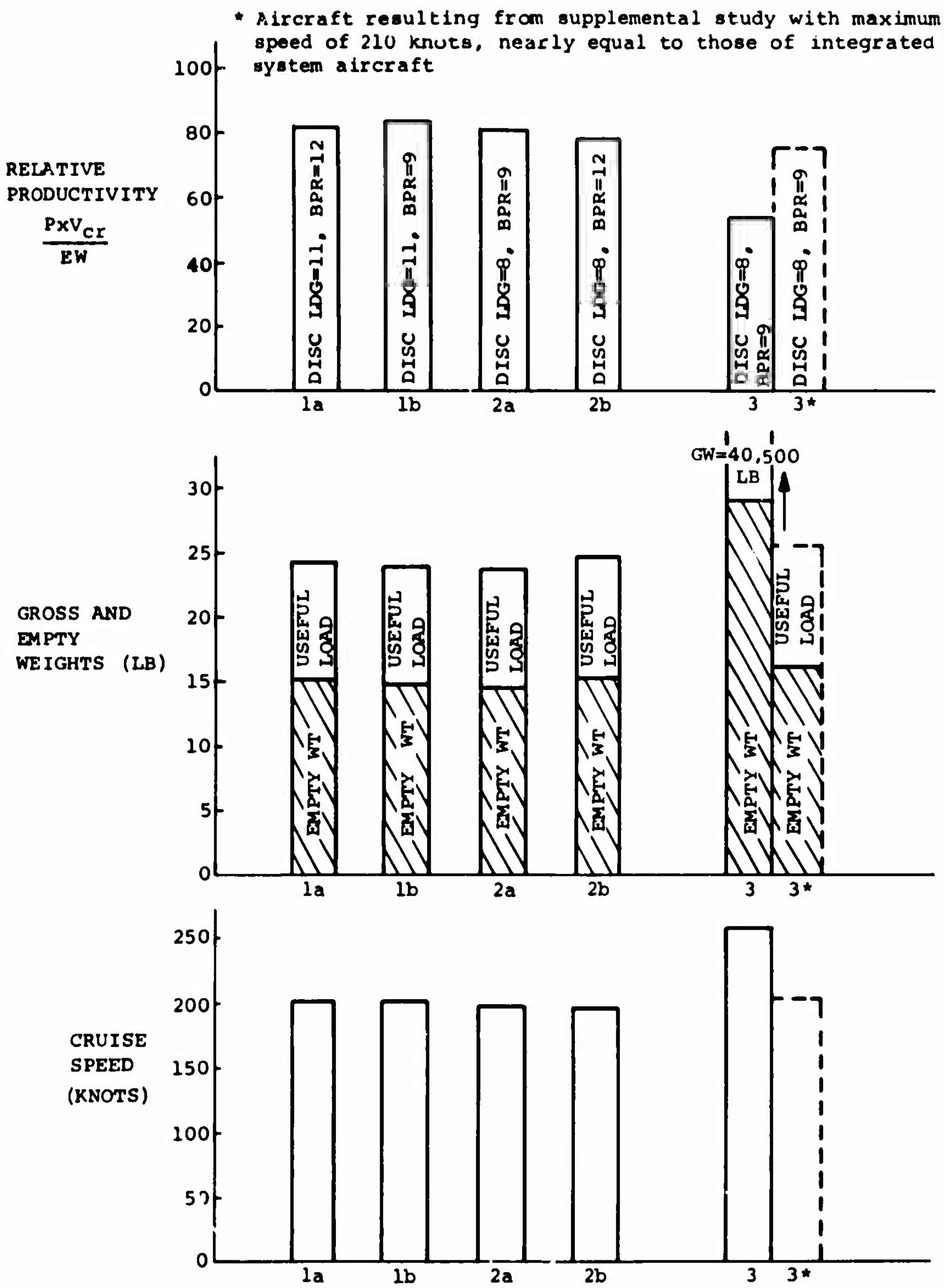


Figure 150. Comparison of Characteristics for Tandem Rotor Propulsion-Unloaded Aircraft

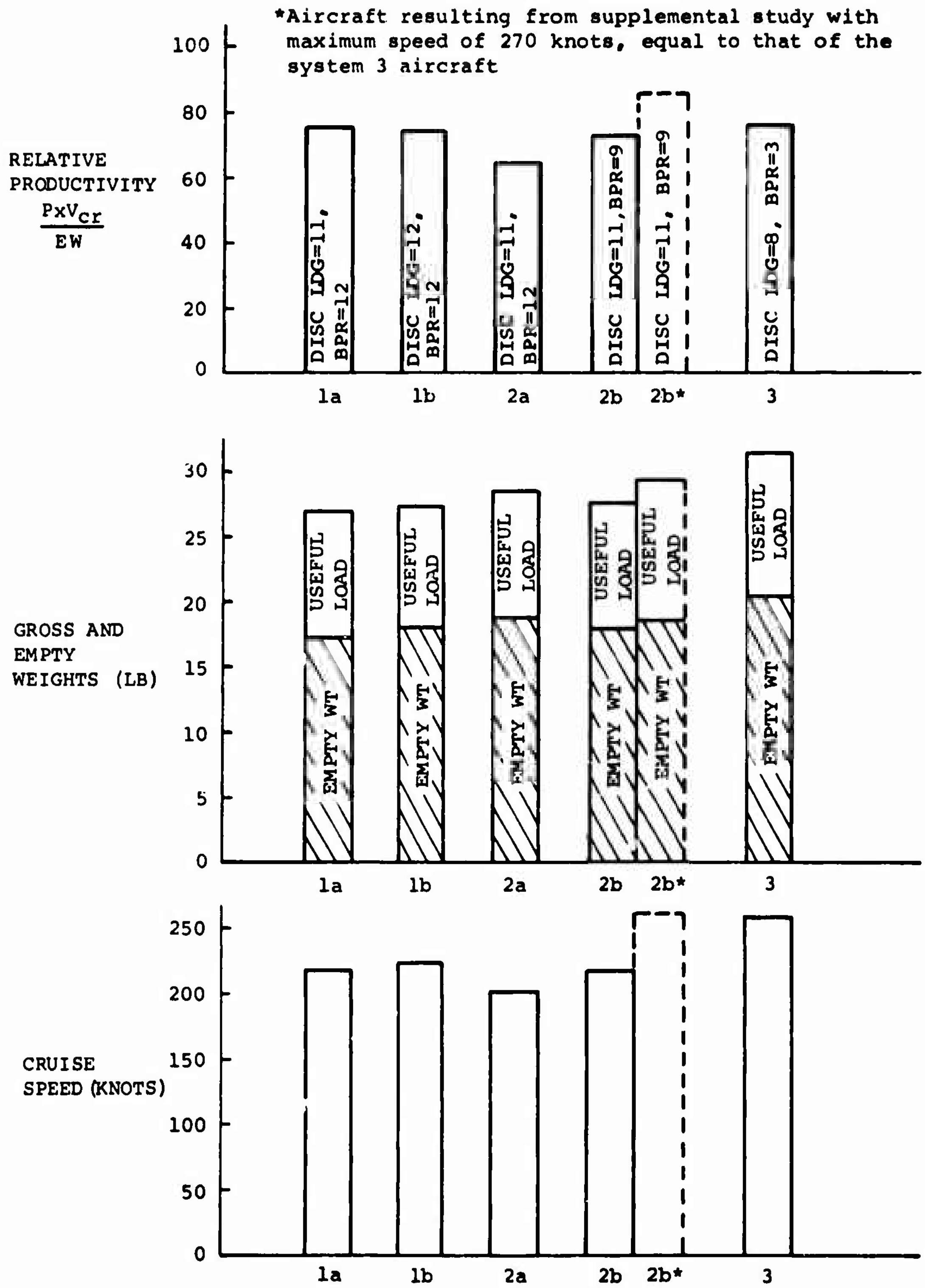


Figure 151. Comparison of Characteristicc for Tandem Rotor Lift/Propulsion-Unloaded Aircraft

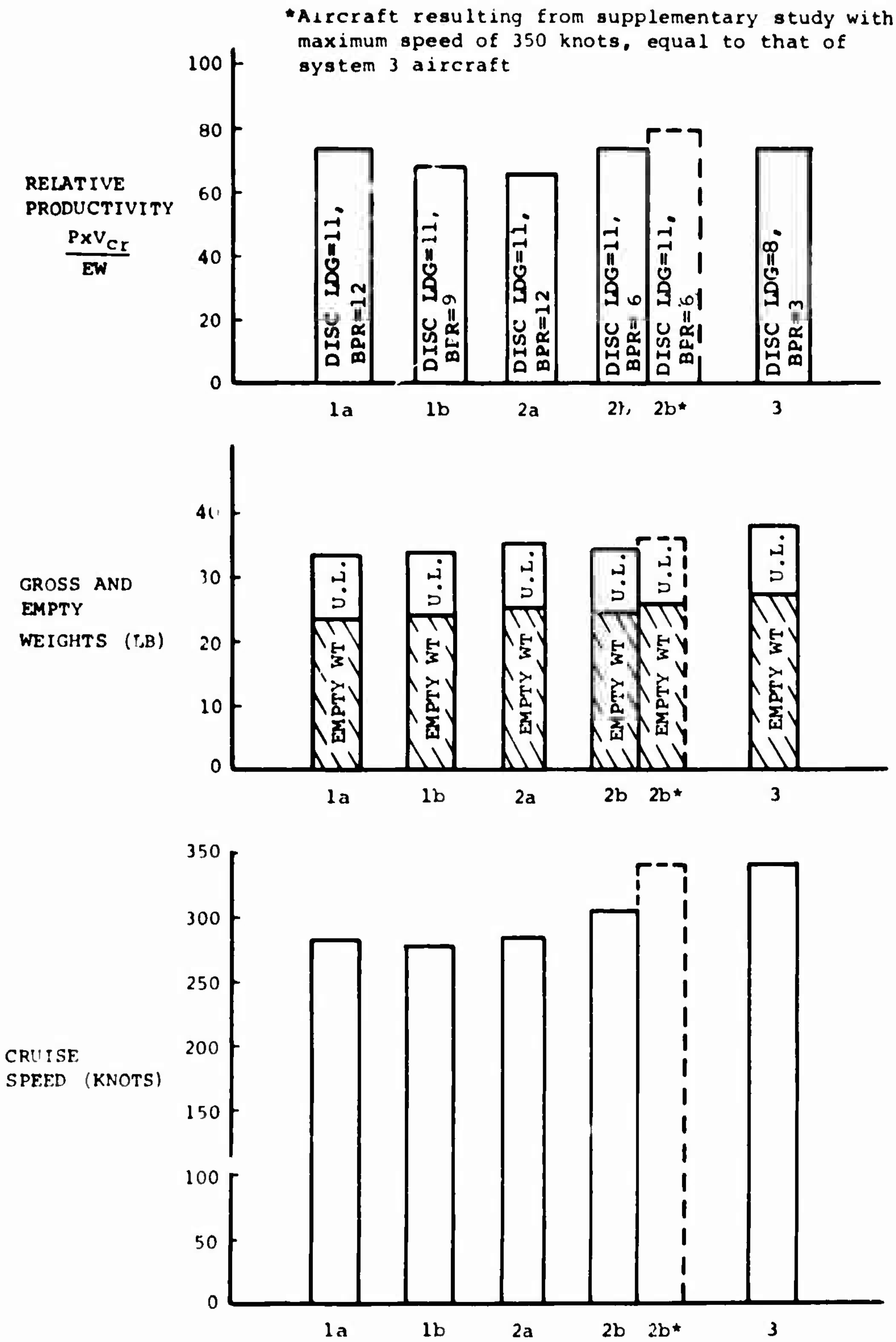


Figure 152. Comparison of Characteristics for Tandem Rotor Composite Aircraft

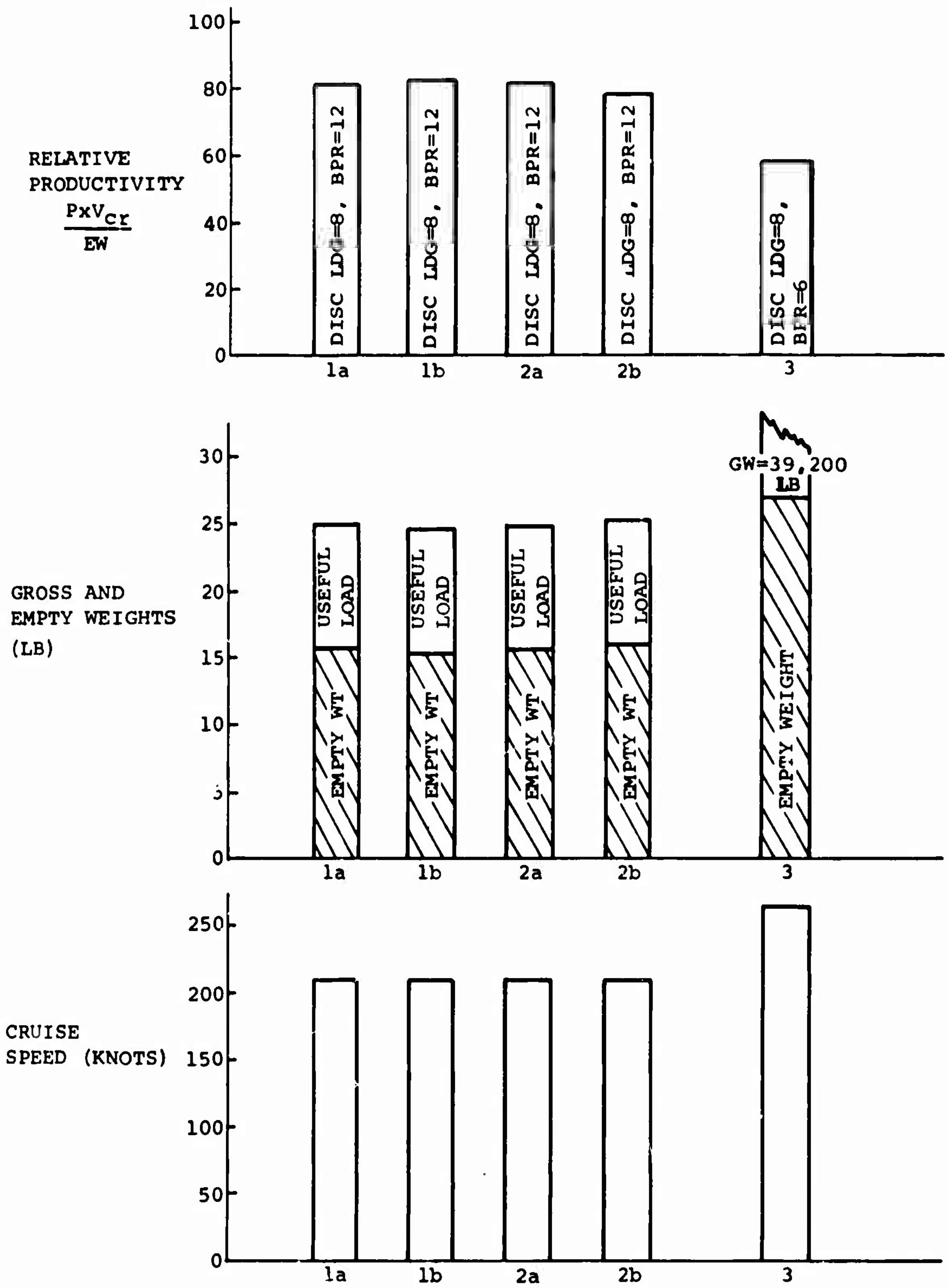


Figure 153. Comparison of Characteristics for Single Rotor Propulsion-Unloaded Aircraft

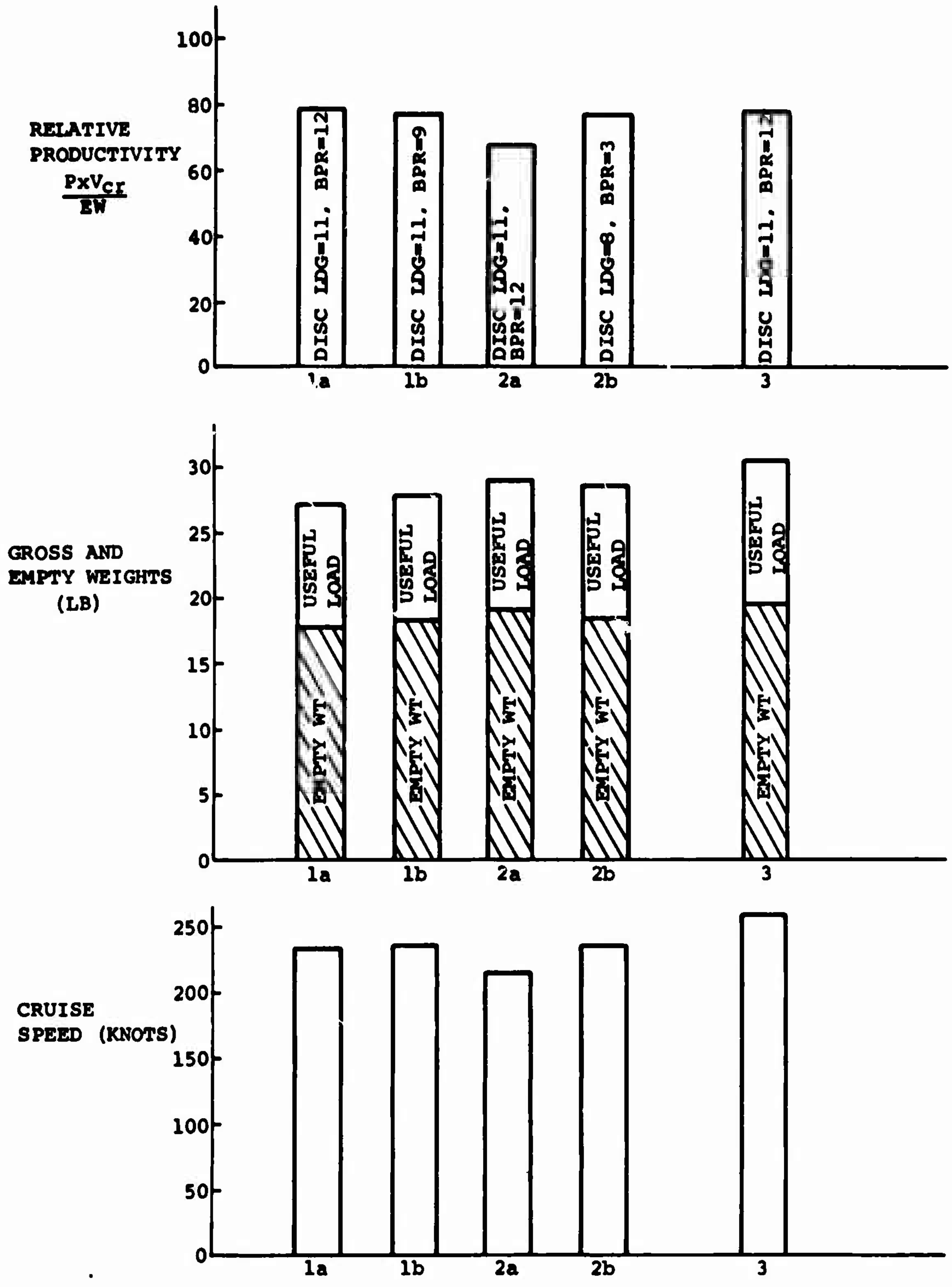


Figure 154. Comparison of Characteristics for Single Rotor Lift/Propulsion-Unloaded Aircraft

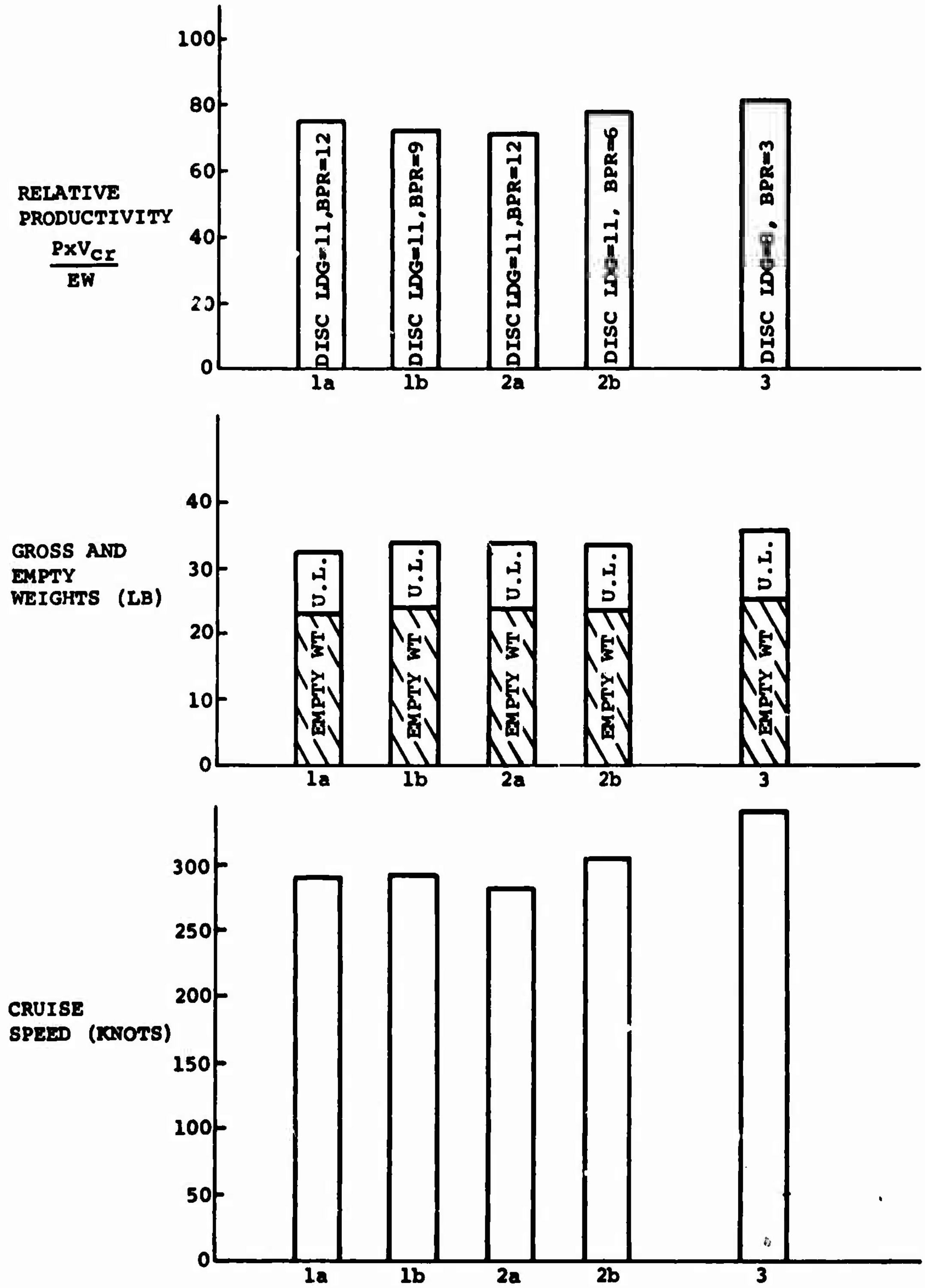


Figure155. Comparison of Characteristics for Single Rotor Composite Aircraft

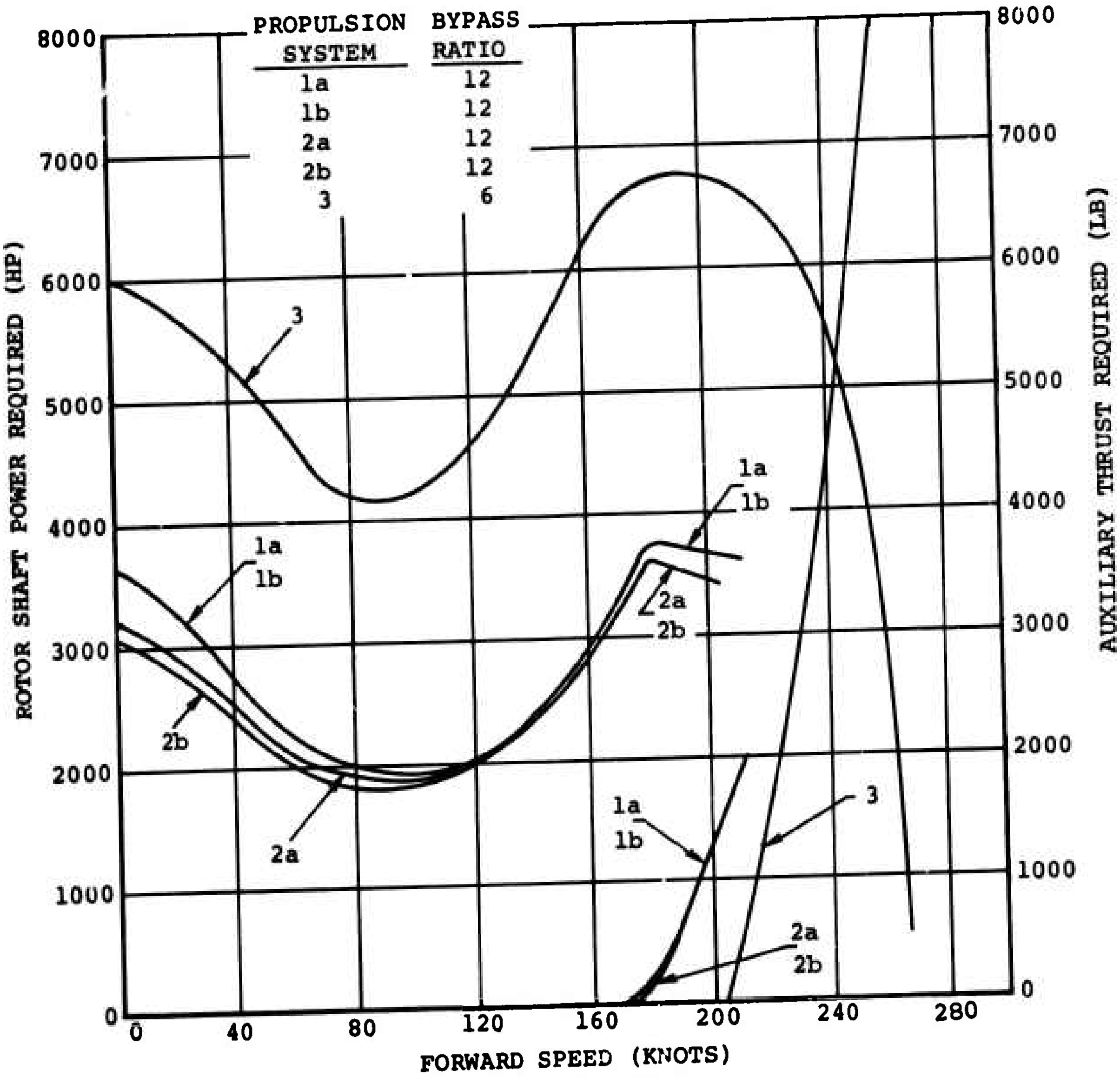


Figure 156. Optimum Tandem Rotor Propulsion-Unloaded Aircraft - Rotor Shaft Power Required and Auxiliary Thrust Required vs. Forward Speed

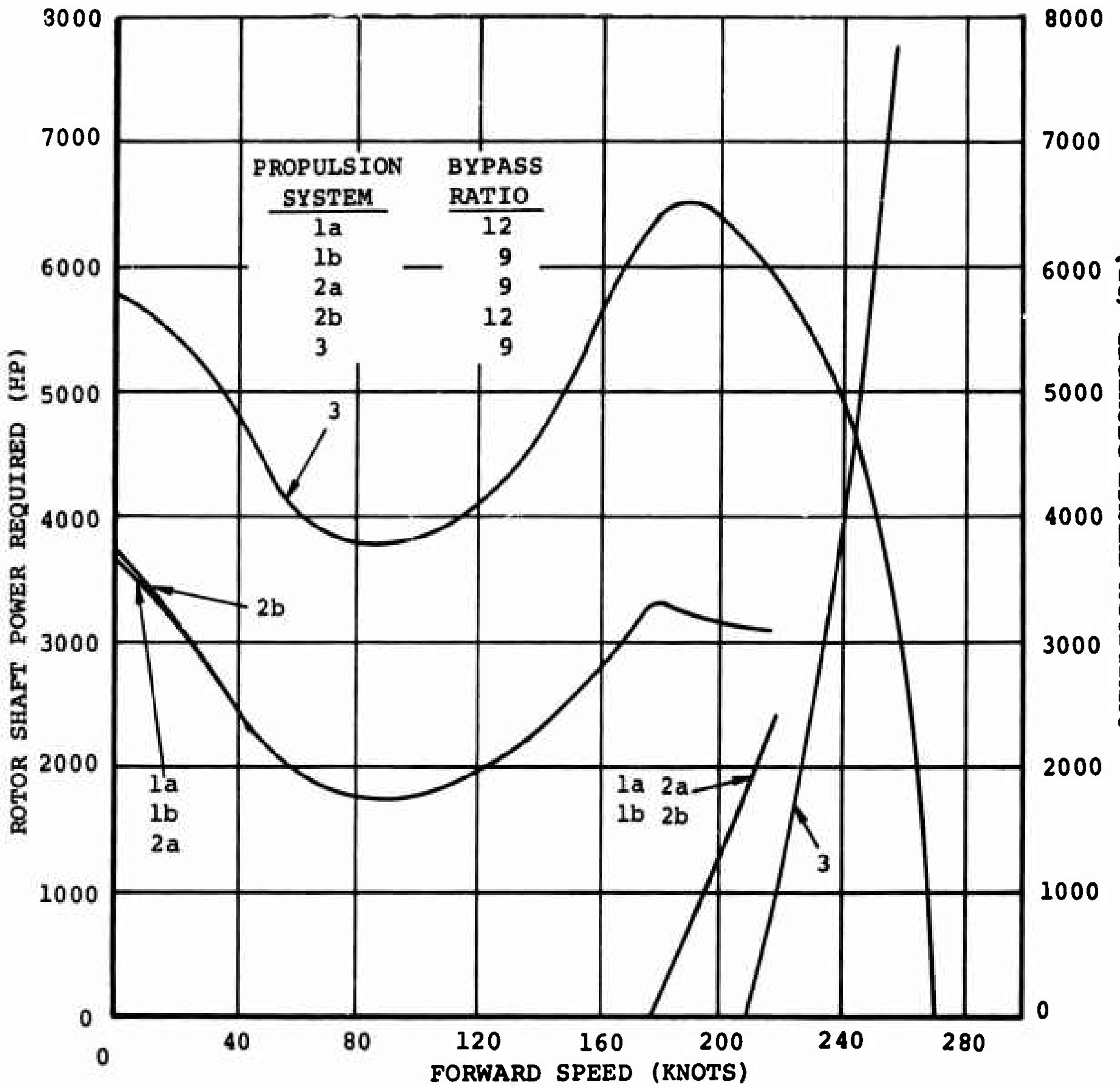


Figure 157. Optimum Single Rotor Propulsion-Unloaded Aircraft - Rotor Shaft Power Required and Auxiliary Thrust Required vs. Forward Speed

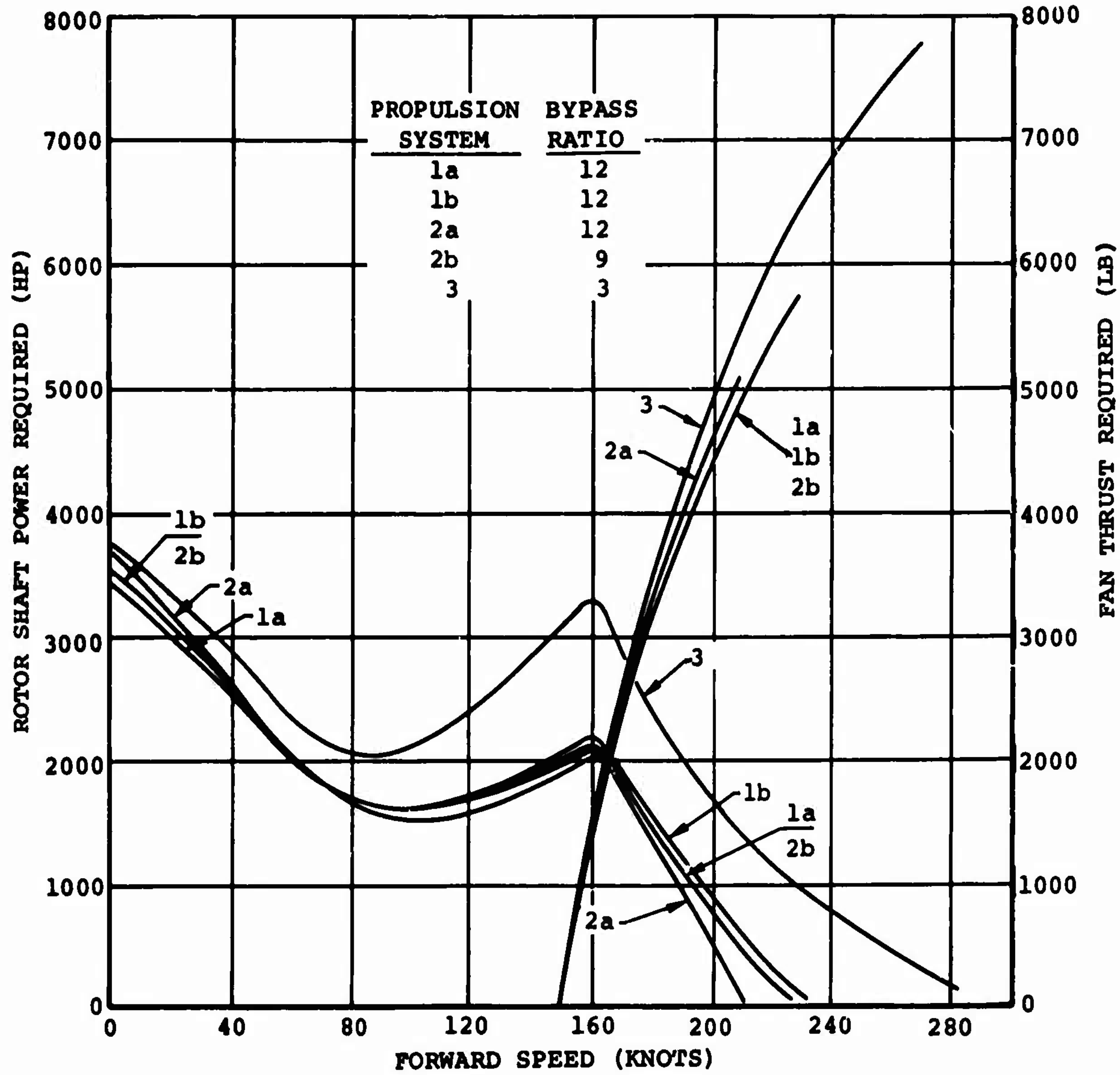


Figure 158. Optimum Tandem Rotor Lift/Propulsion-Unloaded Aircraft - Rotor Shaft Power Required and Fan Thrust Required vs. Forward Speed

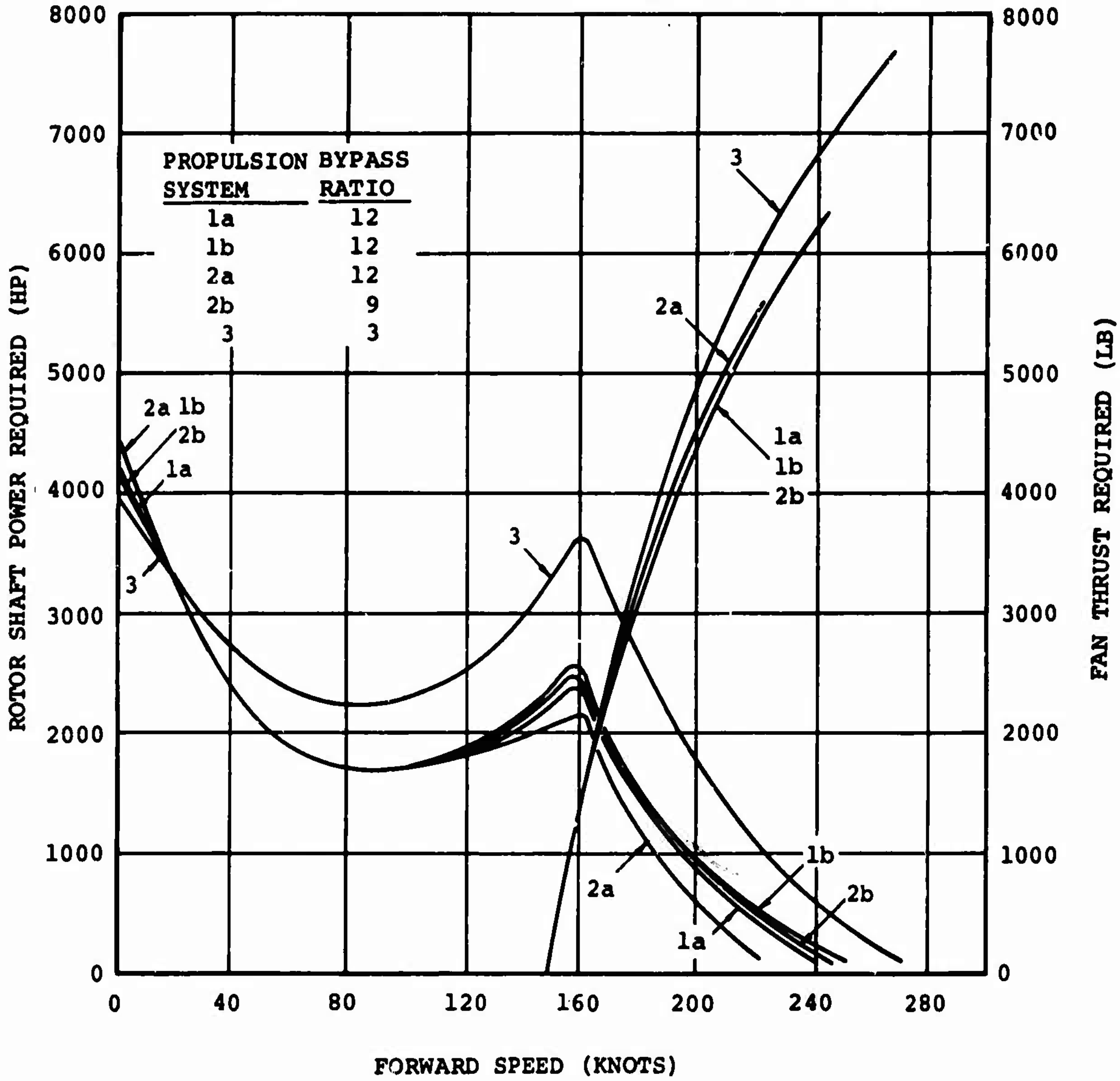


Figure 159. Optimum Single Rotor Lift/Propulsion-Unloaded Aircraft - Rotor Shaft Power Required and Fan Thrust Required vs. Forward Speed

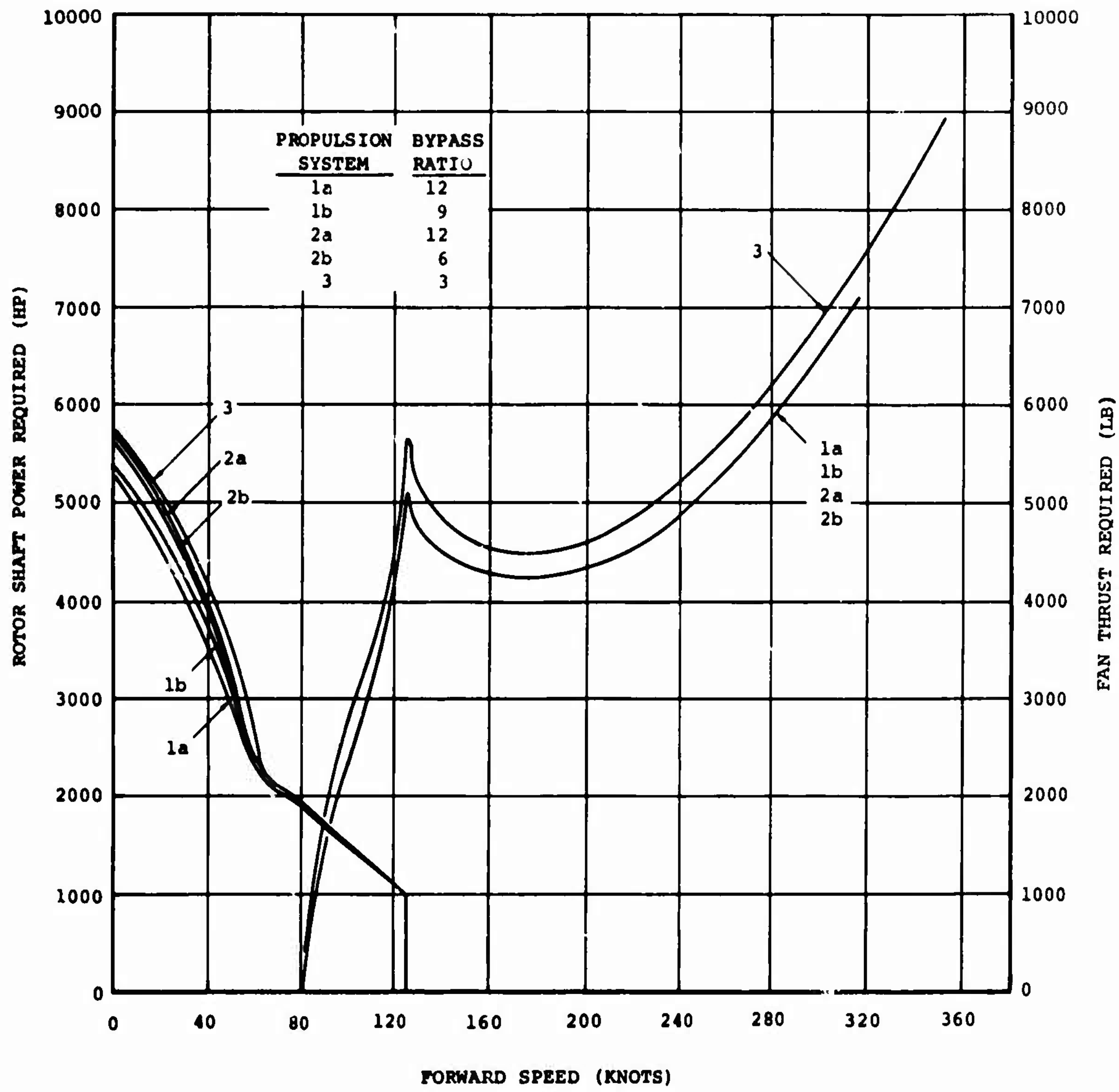


Figure 160. Optimum Tandem Rotor Composite Aircraft - Rotor Shaft Power Required and Fan Thrust Required vs. Forward Speed

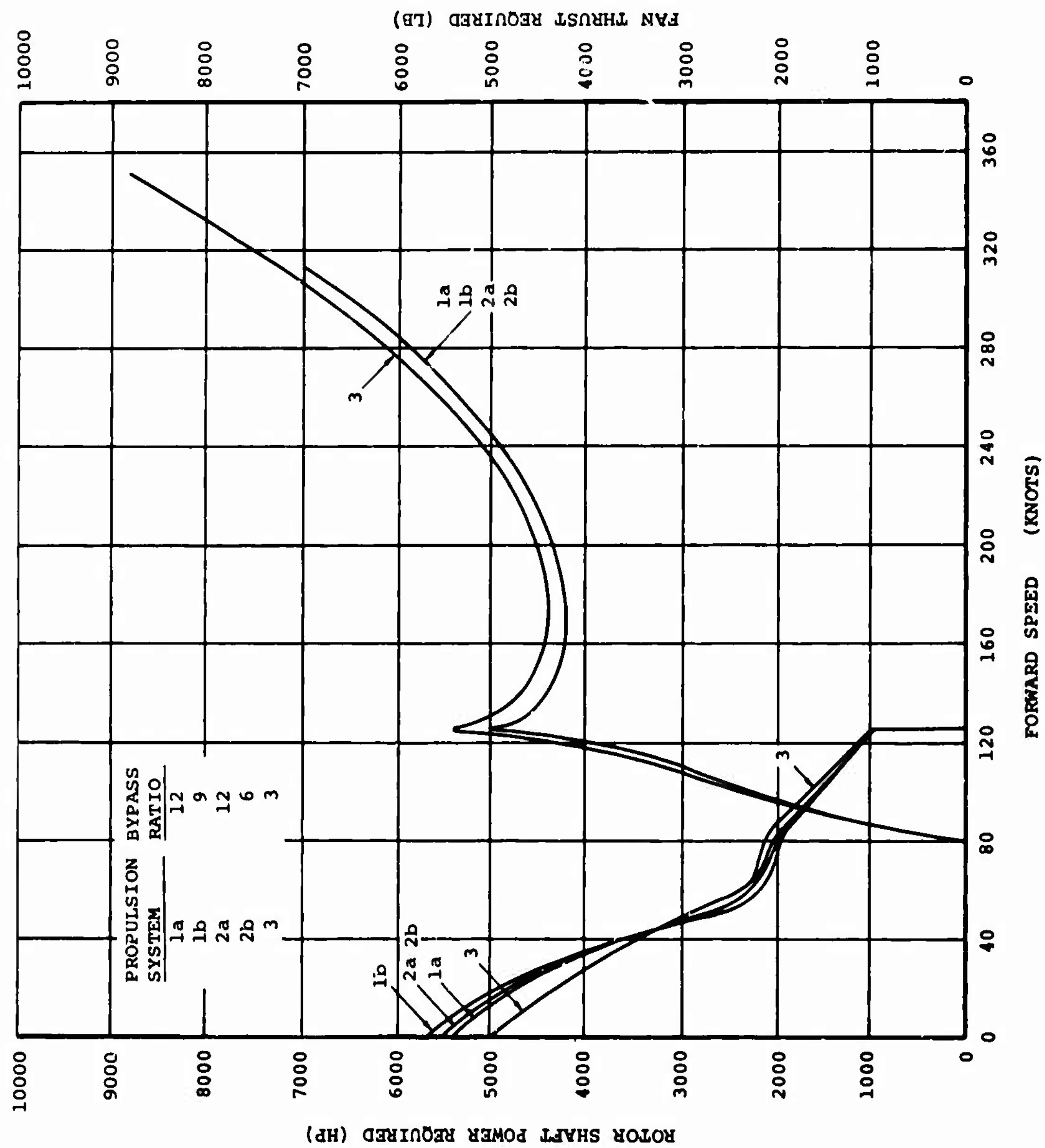


Figure 161. Optimum Single Rotor Composite Aircraft - Rotor Shaft Power Required and Fan Thrust Required vs. Forward Speed

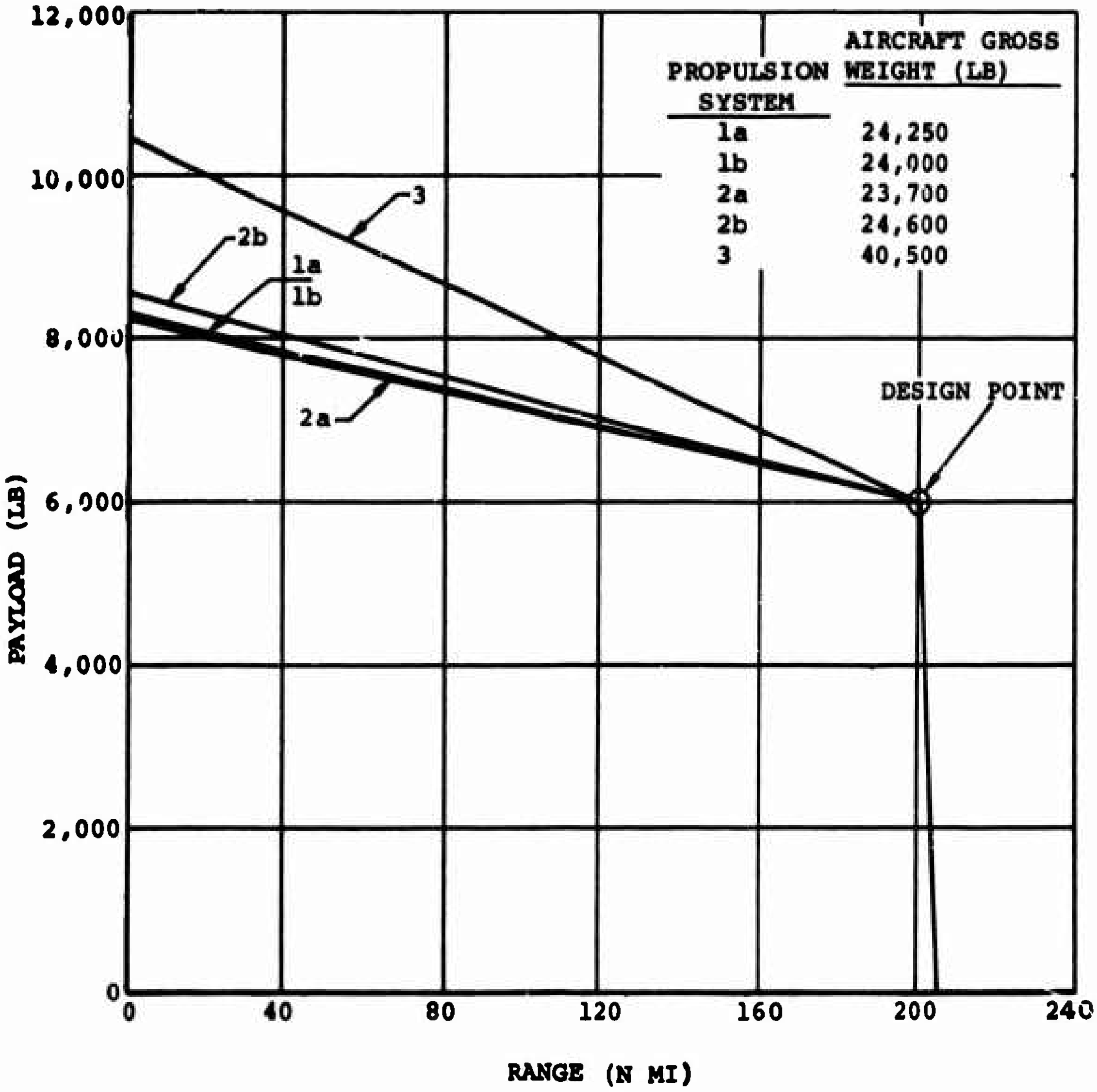


Figure 162. Payload Variation With Range of Tandem Rotor Propulsion-Unloaded Aircraft

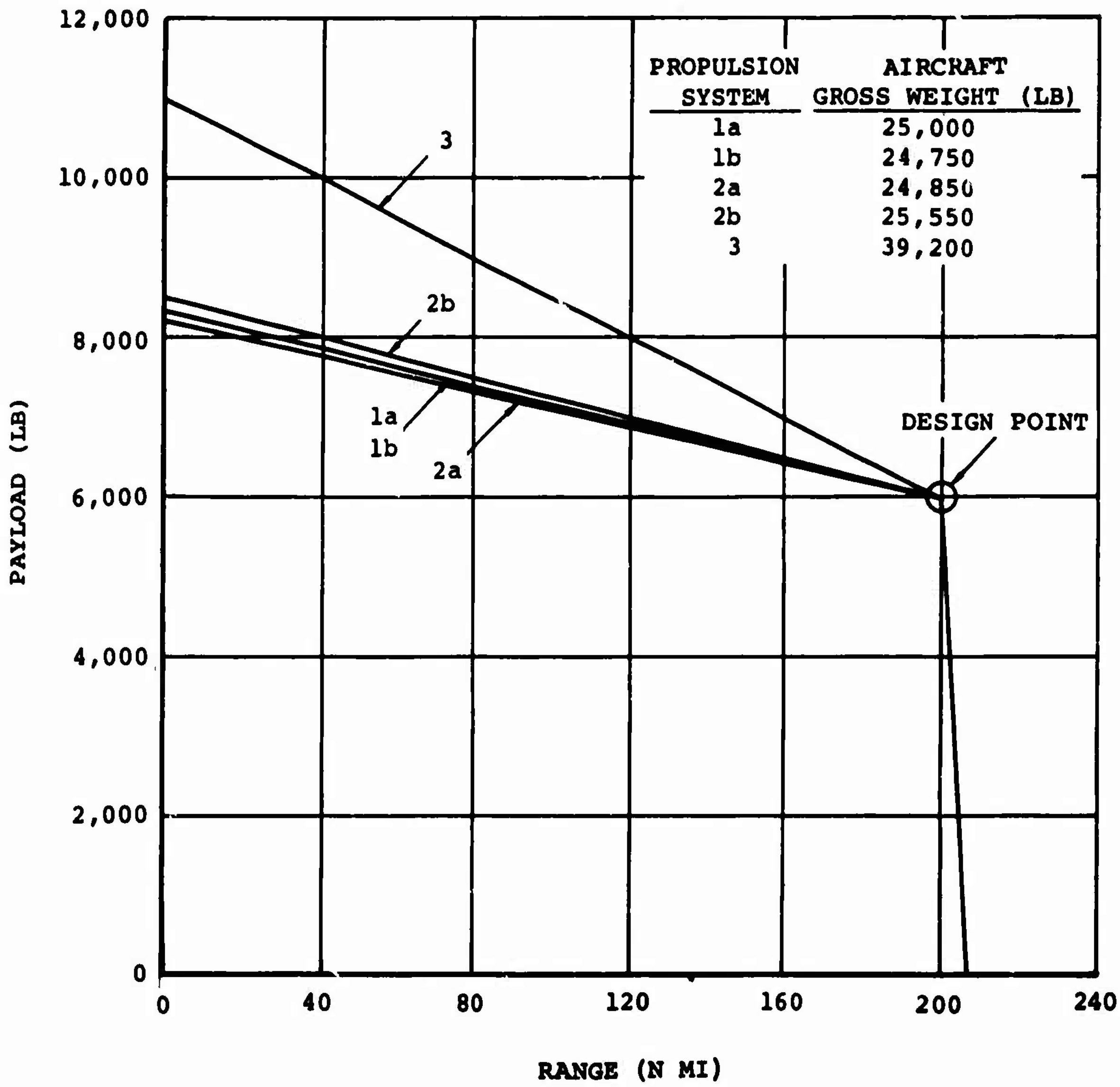


Figure 163. Payload Variation With Range of Single Rotor Propulsion-Unloaded Aircraft

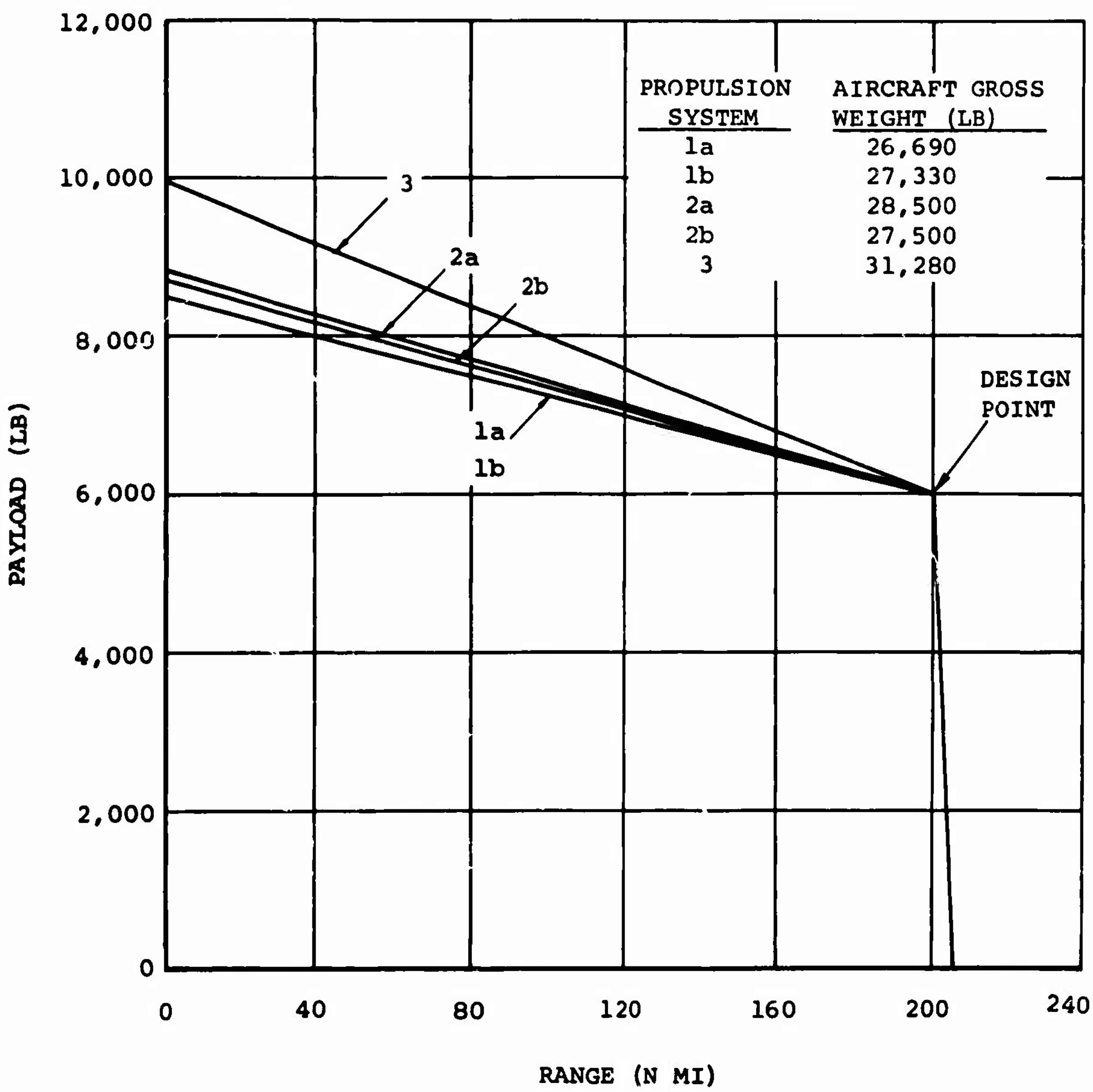


Figure 164. Payload Variation With Range of Tandem Rotor Lift/Propulsion-Unloaded Aircraft

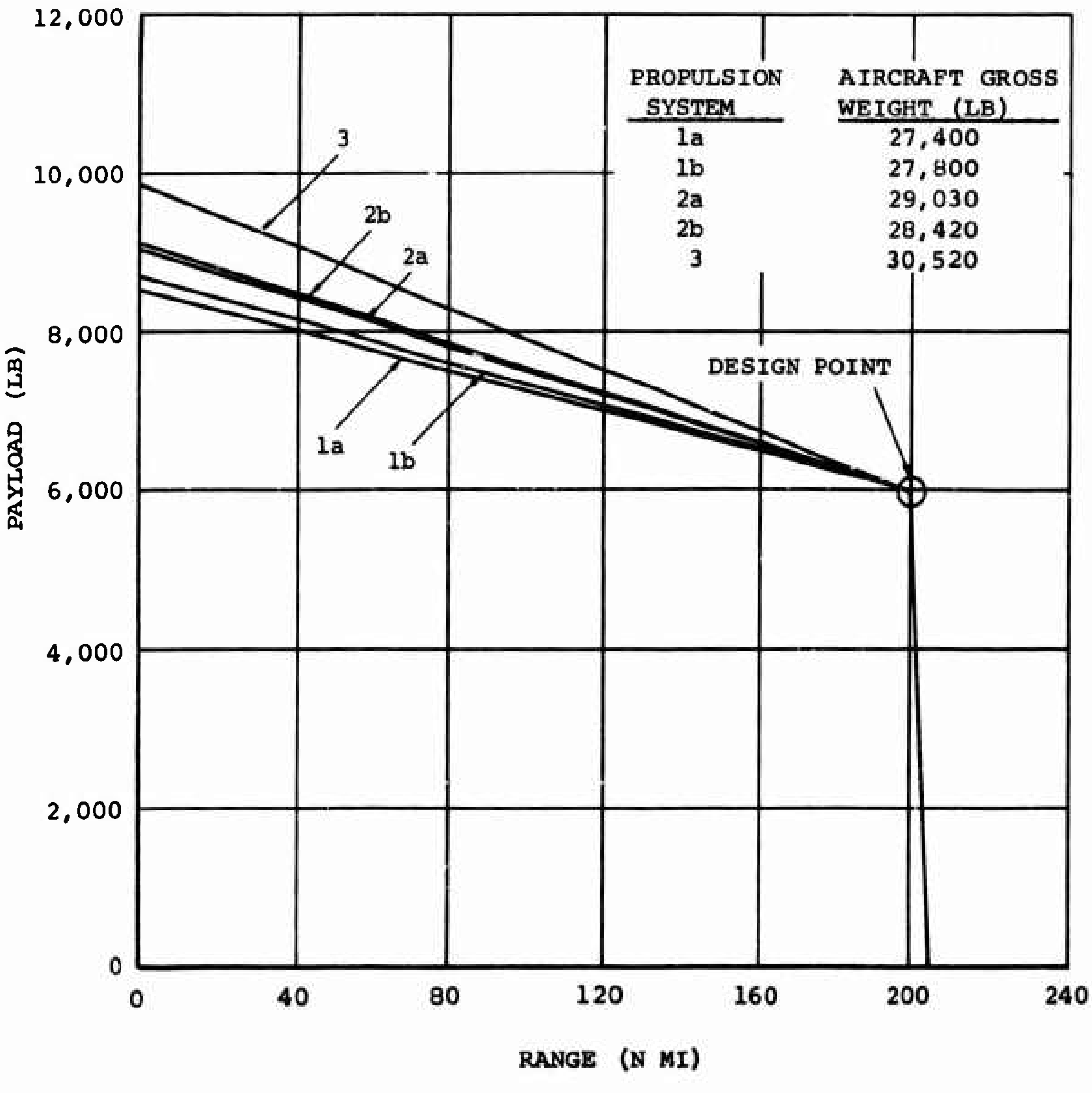


Figure 165. Payload Variation With Range of Single Rotor Lift/Propulsion-Unloaded Aircraft

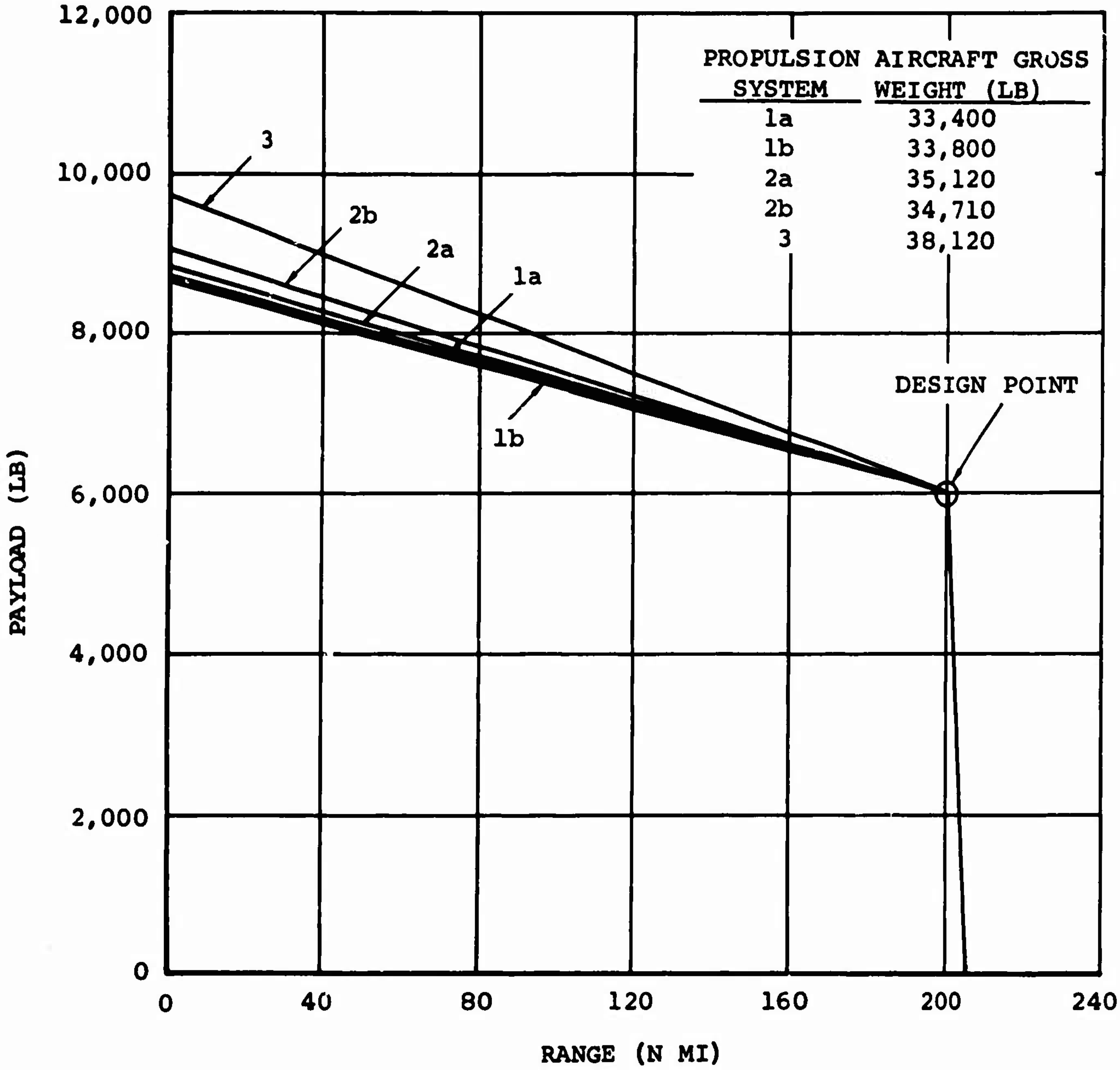


Figure 166. Payload Variation With Range of Tandem Rotor Composite Aircraft

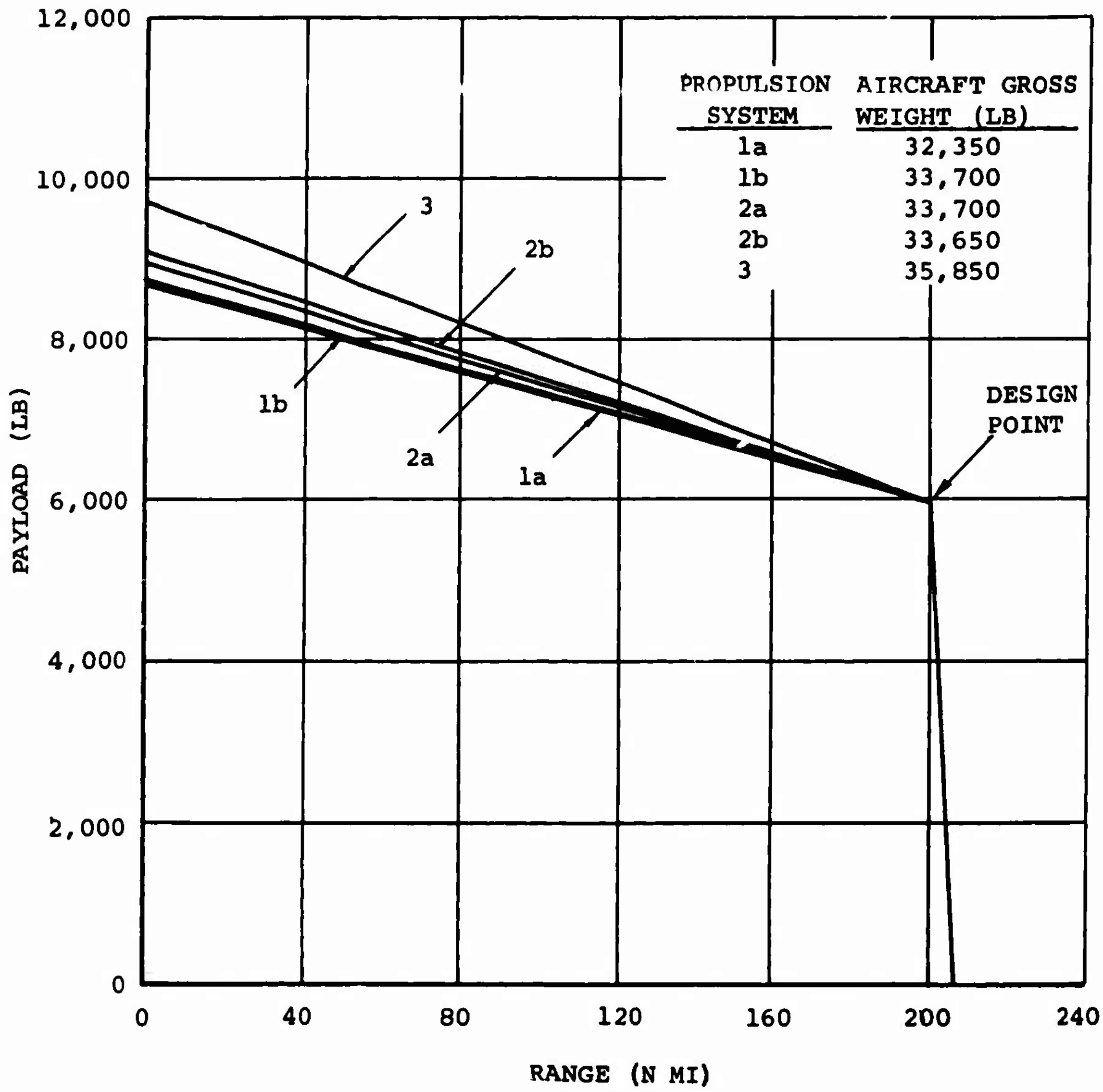


Figure 167. Payload Variation With Range of Single Rotor Composite Aircraft

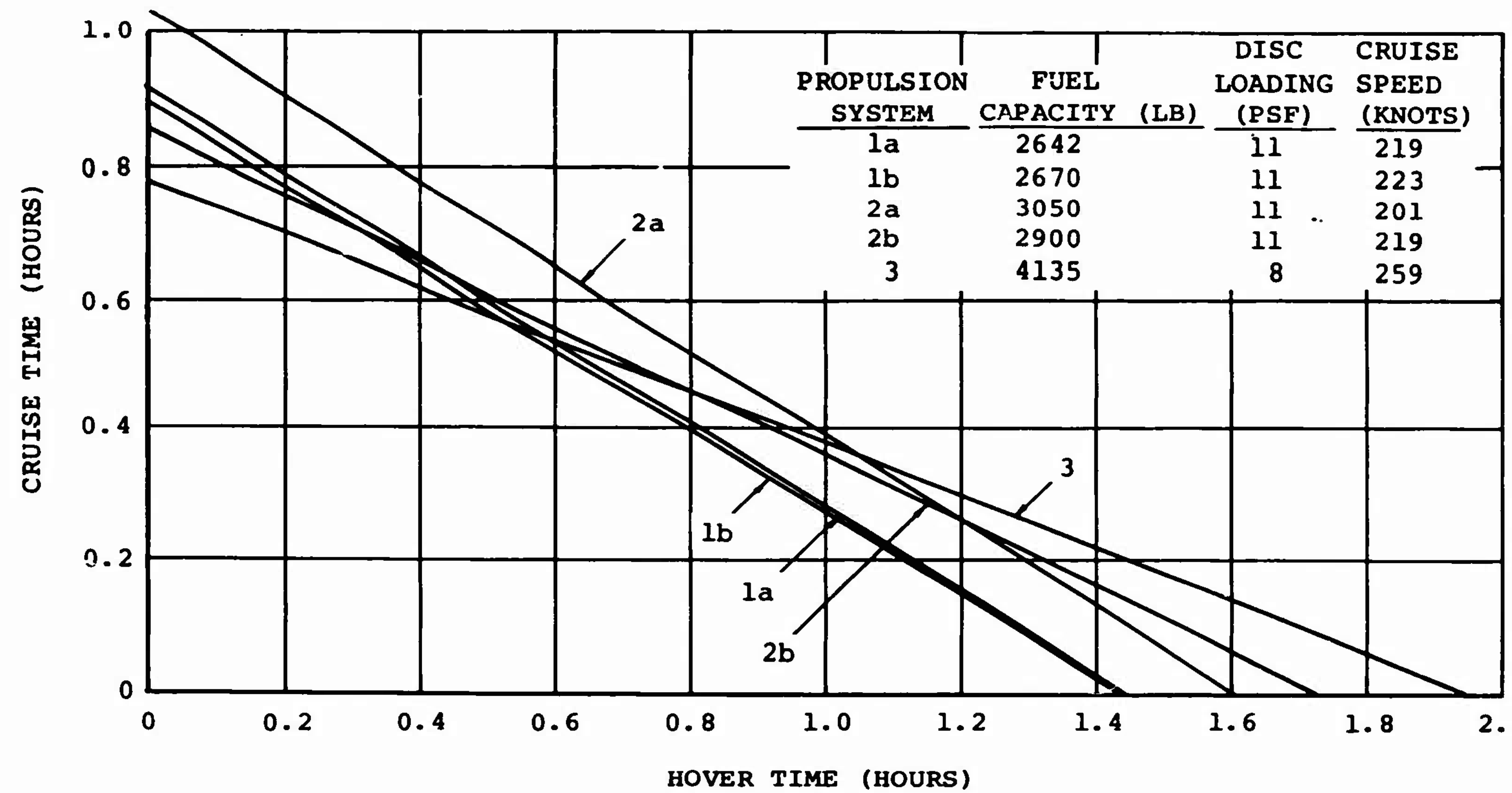


Figure 168. Tradeoff of Cruise Time With Hover Time for Tandem Rotor Propulsion-Unloaded Aircraft

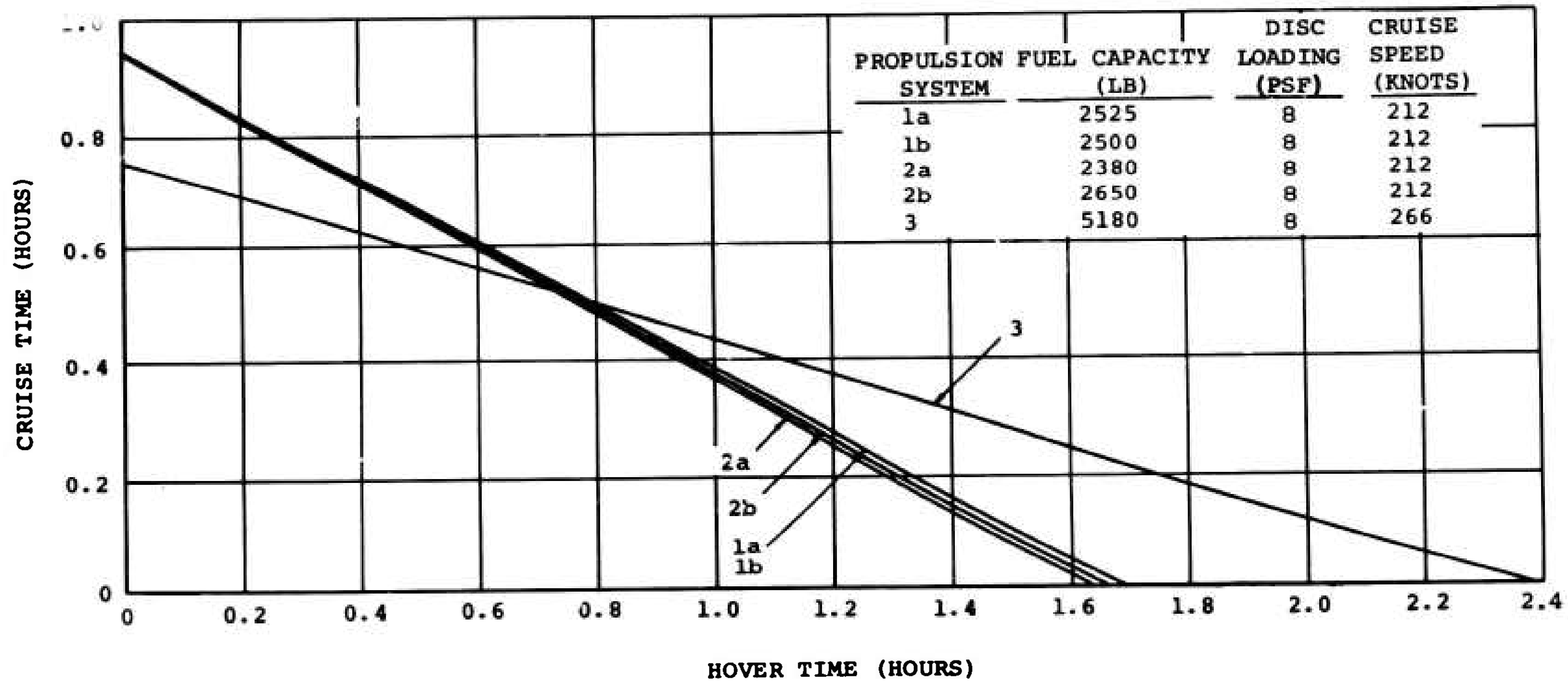


Figure 169. Tradeoff of Cruise Time With Hover Time for Single Rotor Propulsion-Unloaded Aircraft

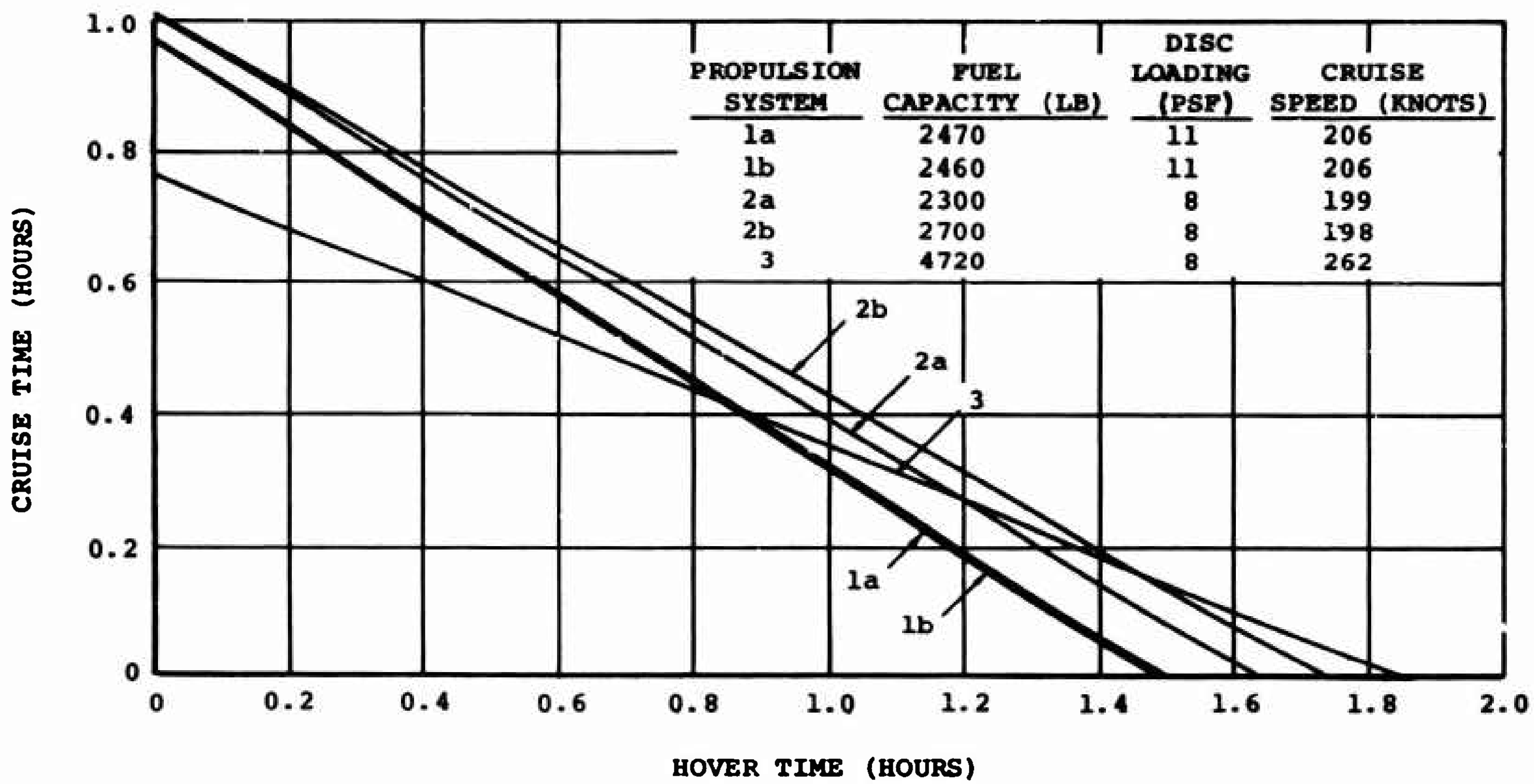


Figure 170. Tradeoff of Cruise Time With Hover Time for Tandem Rotor Lift/Propulsion-Unloaded Aircraft

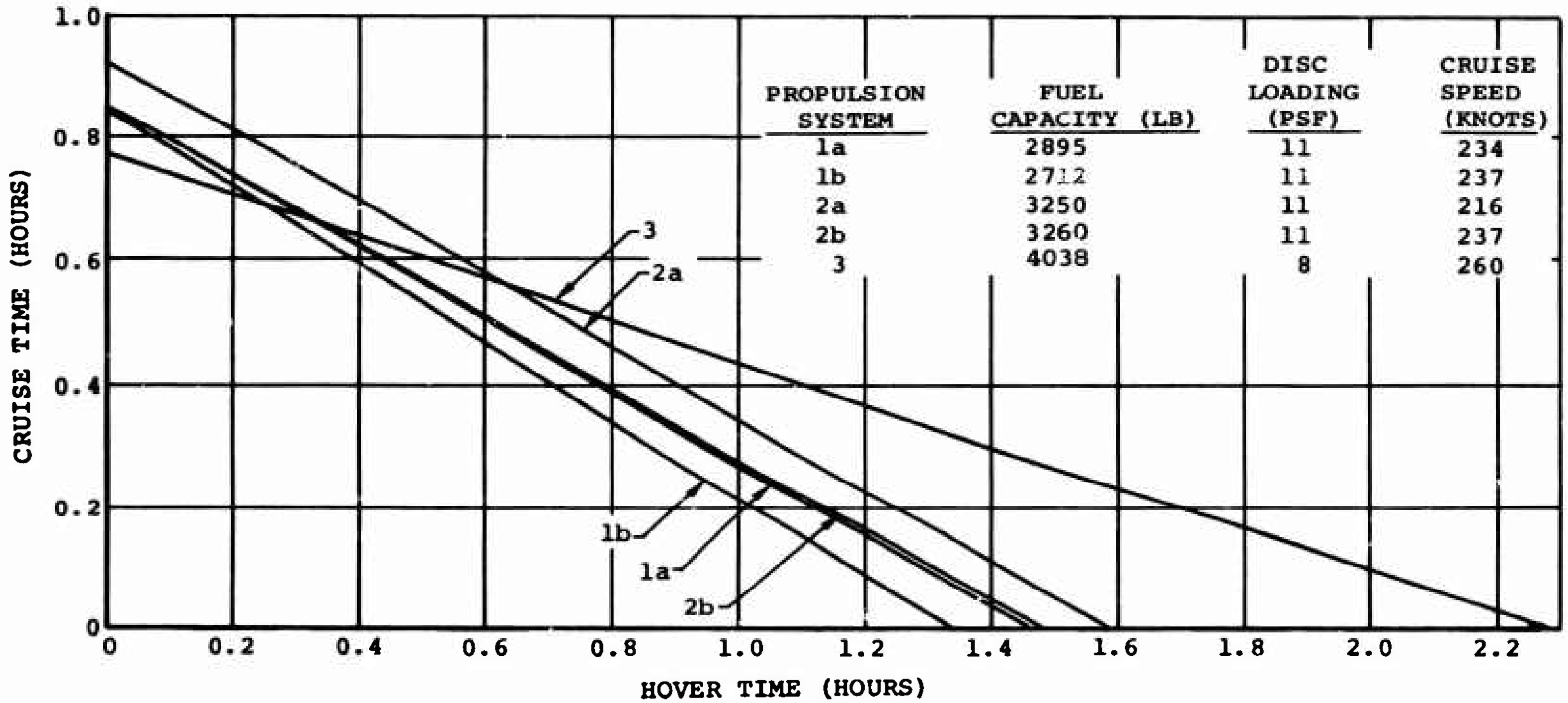


Figure 171. Tradeoff of Cruise Time With Hover Time for Single Rotor Lift/Propulsion-Unloaded Aircraft

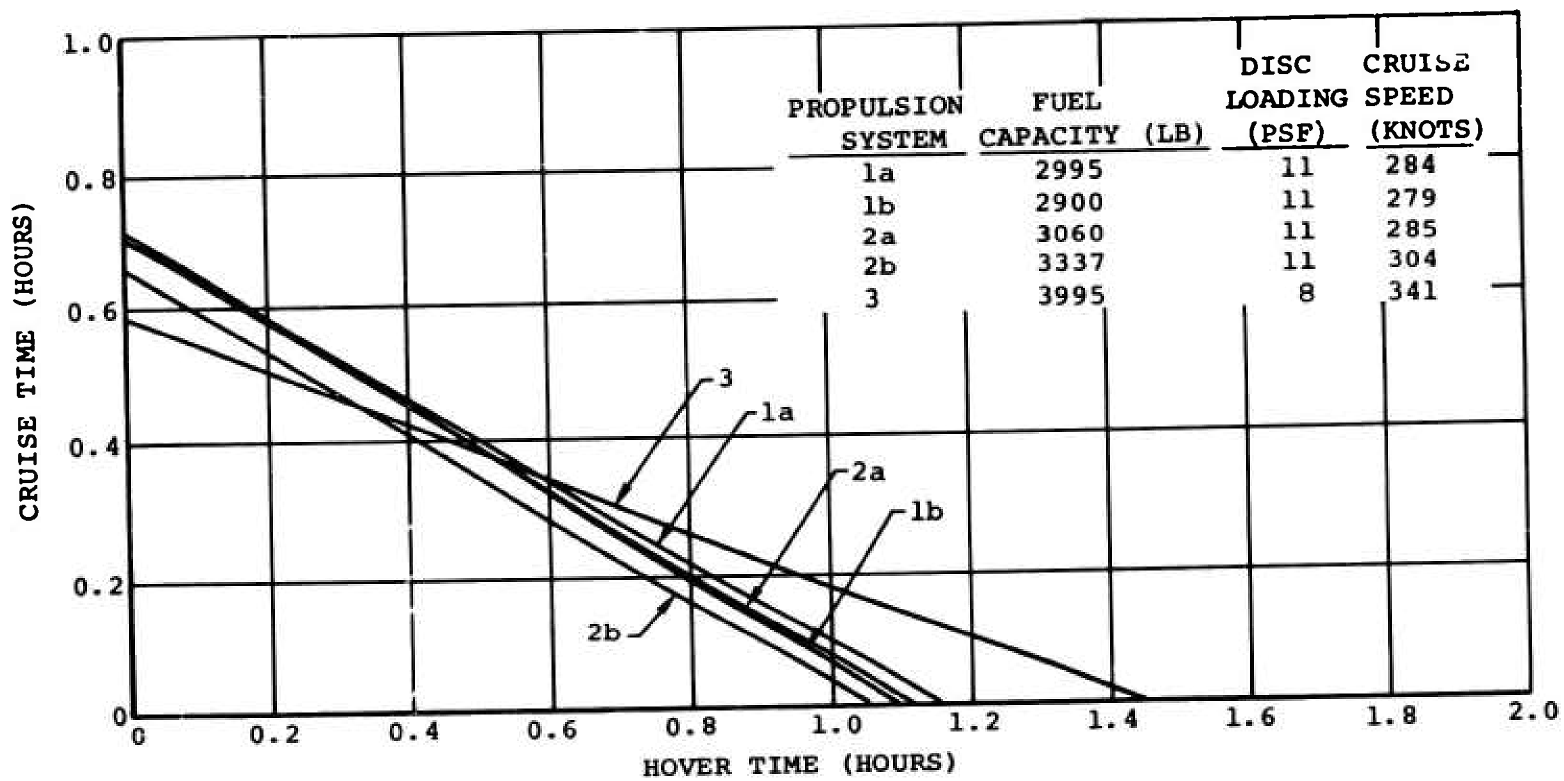


Figure 172. Tradeoff of Cruise Time With Hover Time for Tandem Rotor Composite Aircraft

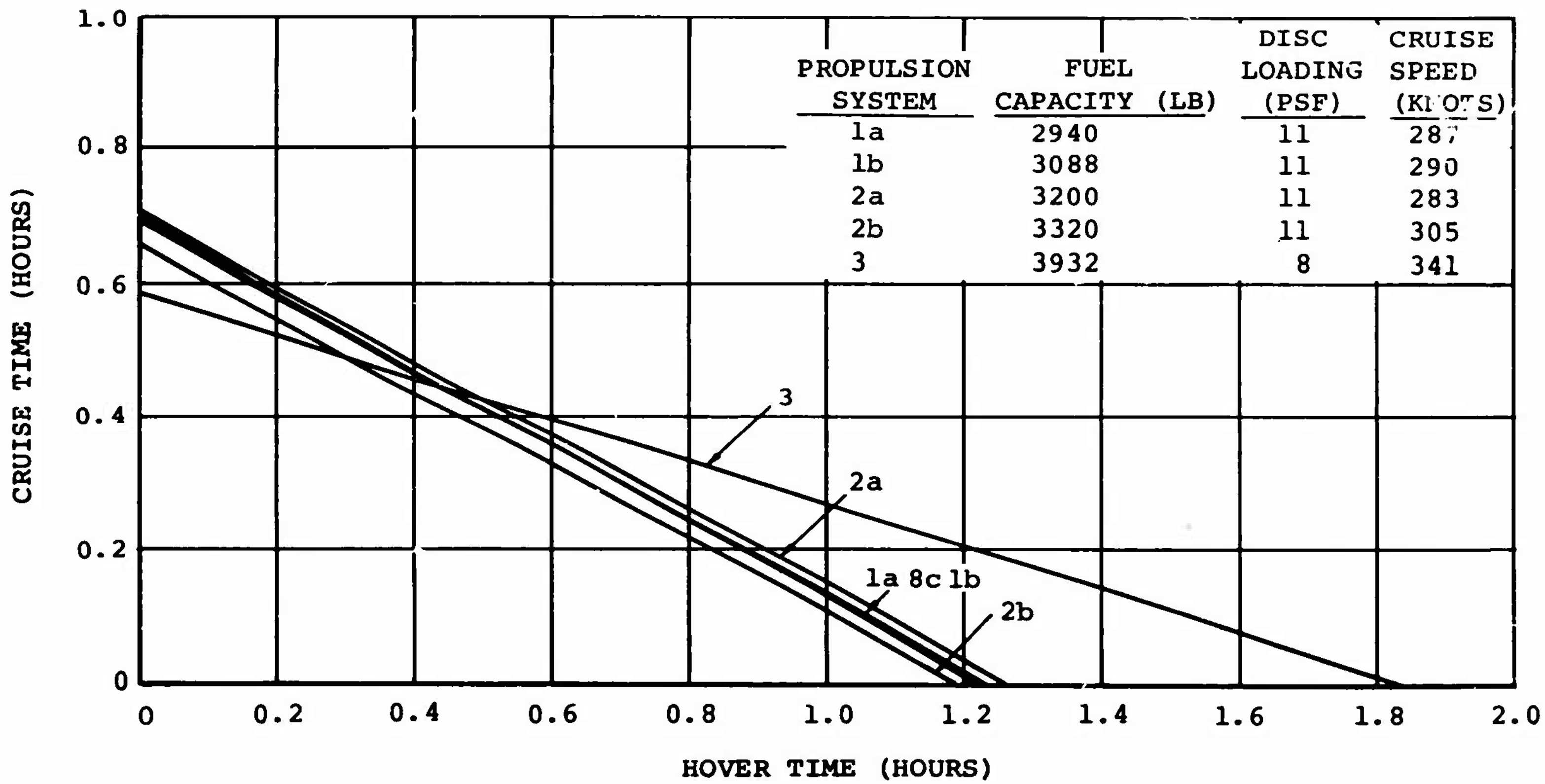


Figure 173. Tradeoff of Cruise Time With Hover Time for Single Rotor Composite Aircraft

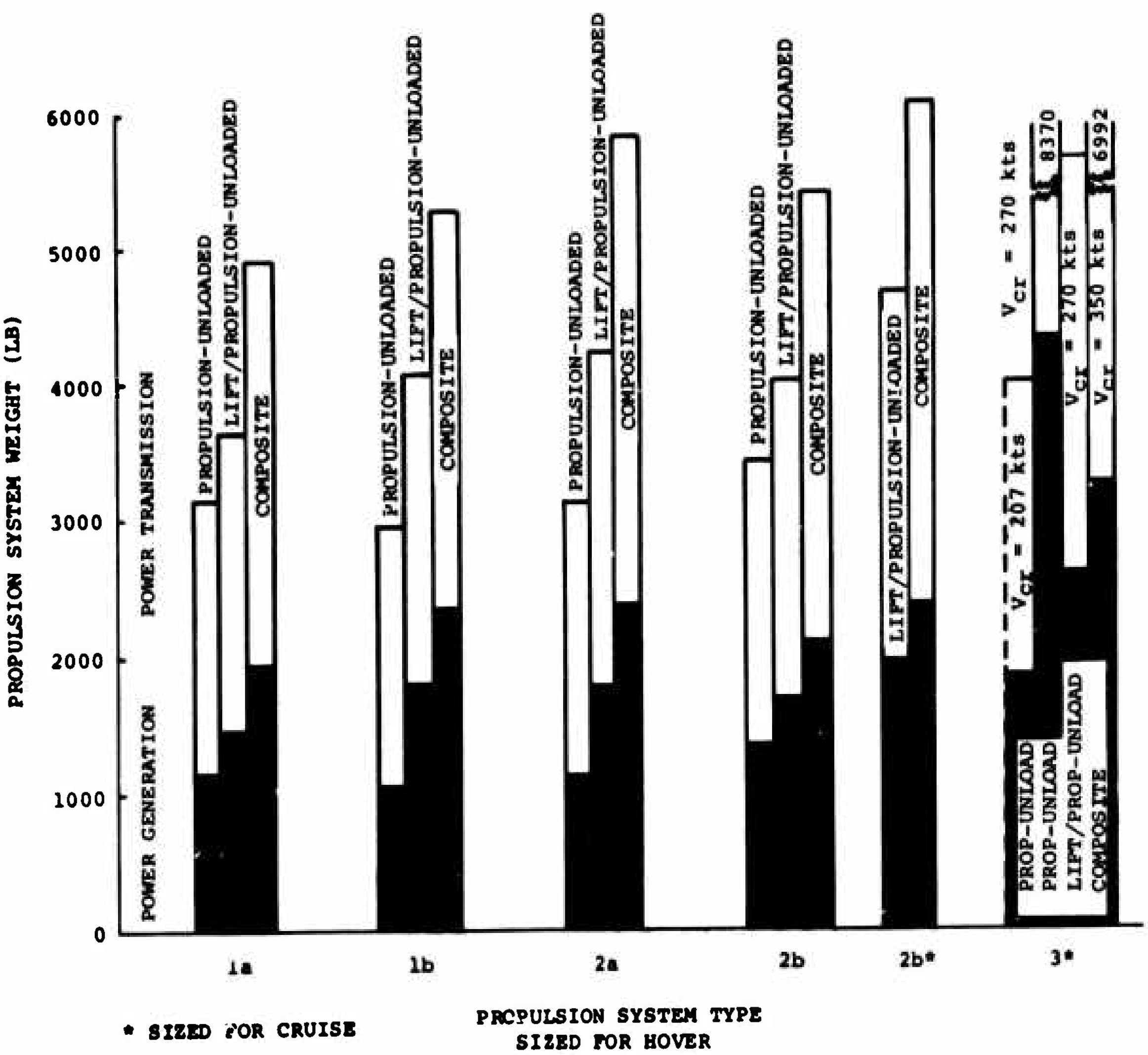


Figure 174. Propulsion System Weight Comparisons of Tandem Rotor Aircraft

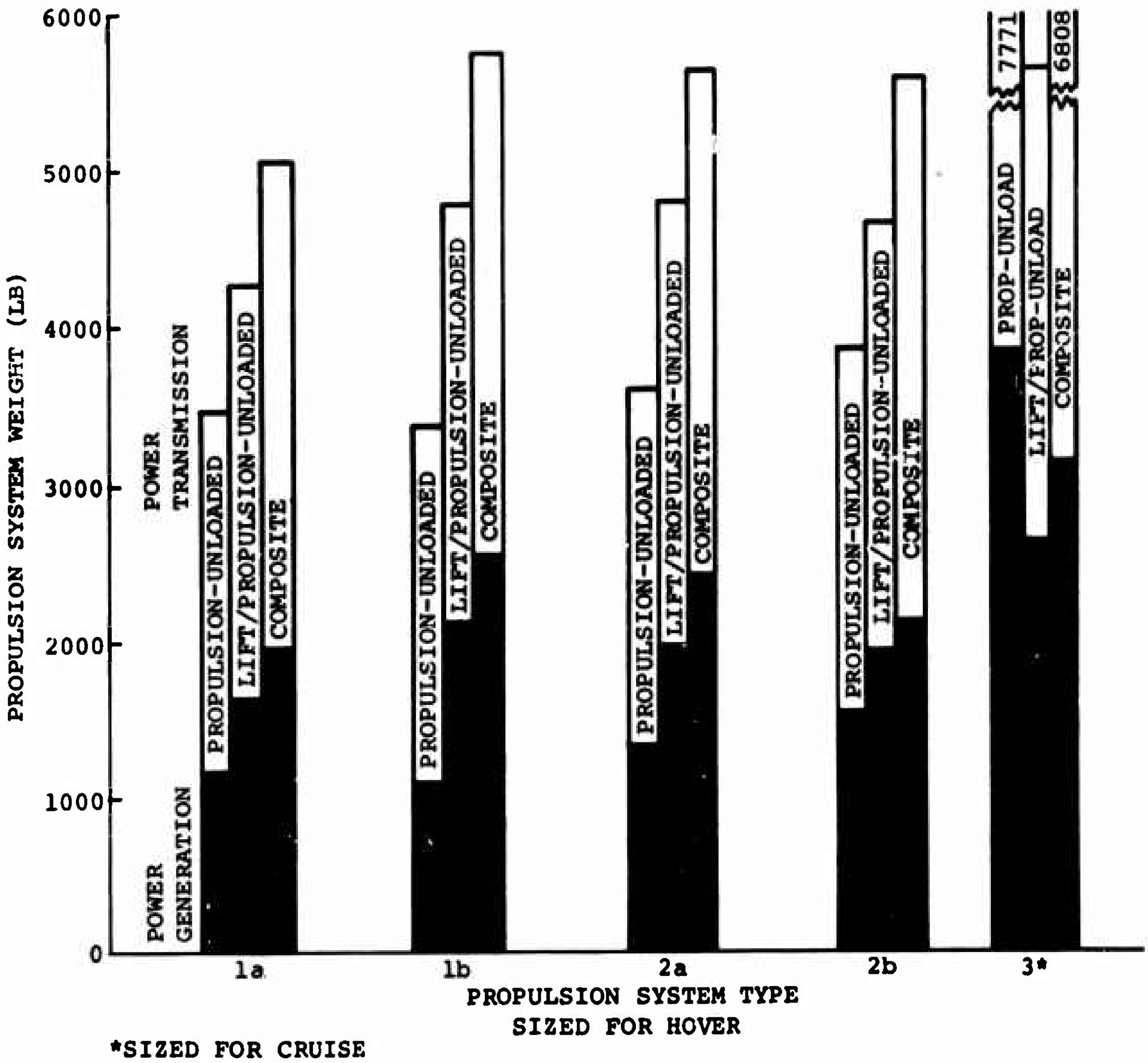


Figure 175. Propulsion System Weight Comparisons of Single Rotor Aircraft

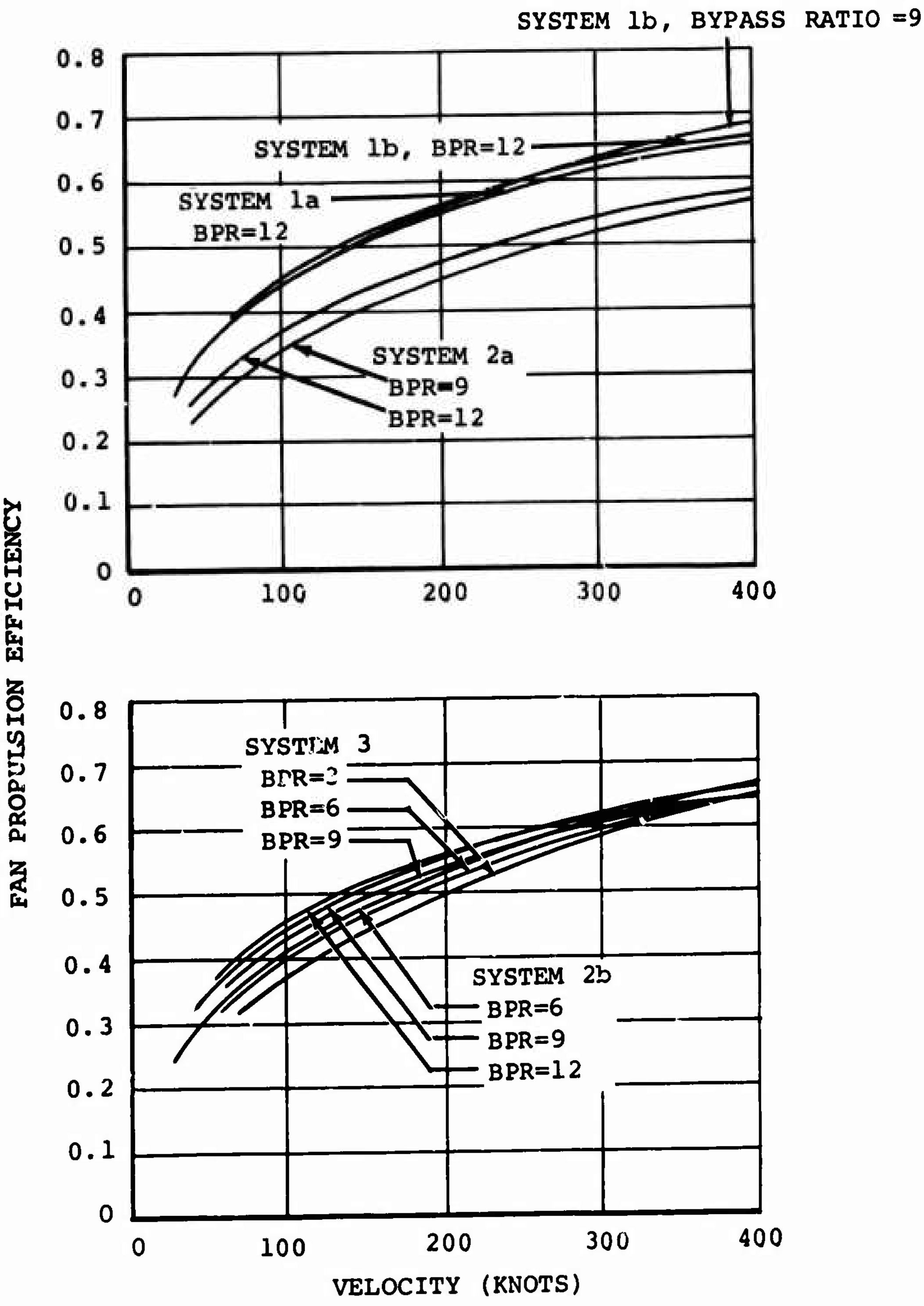


Figure 176. Propulsion Efficiency of Fans

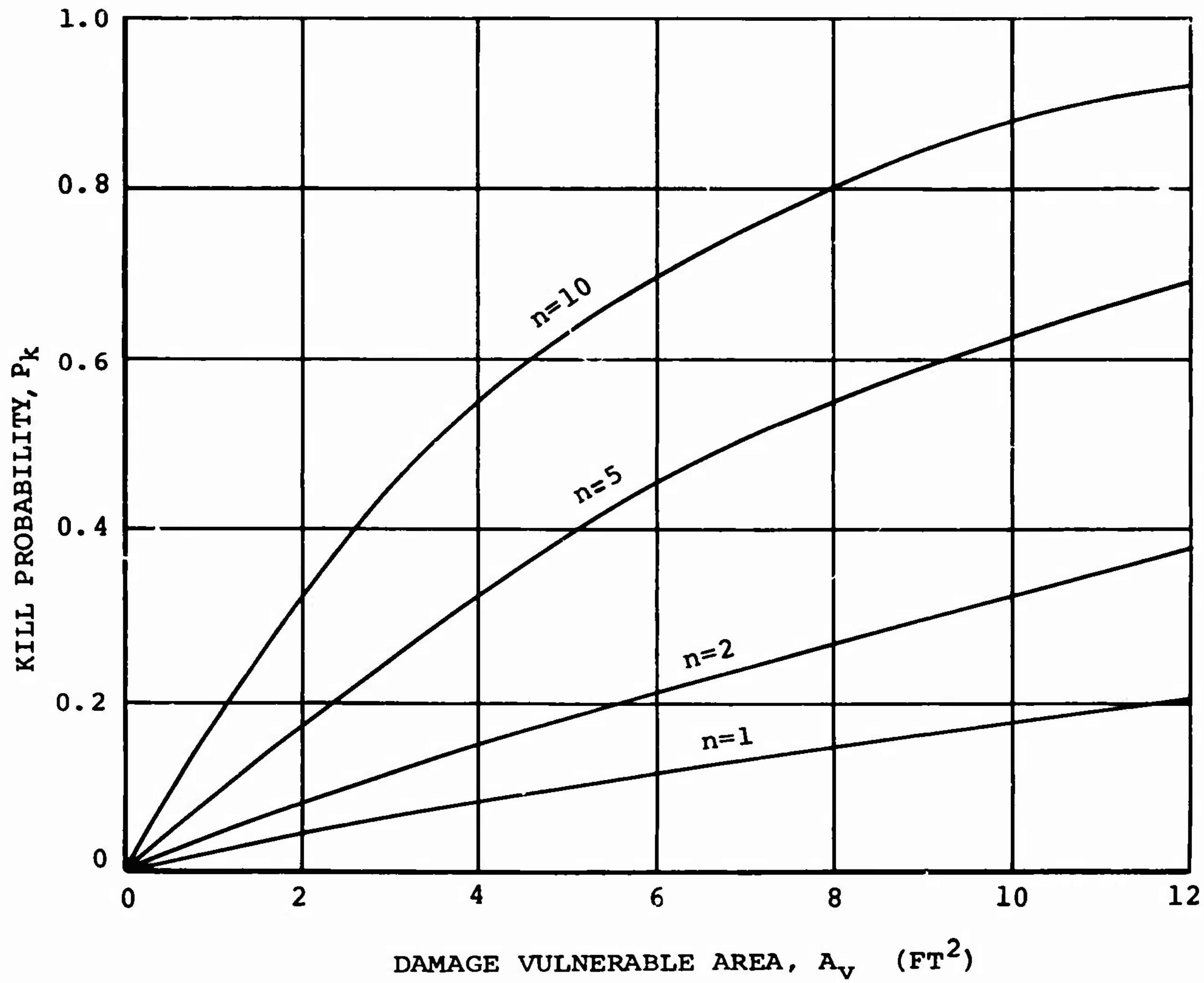


Figure 177. Example of Kill Probability vs. Damage Vulnerable Area for a Given Probability of Hit

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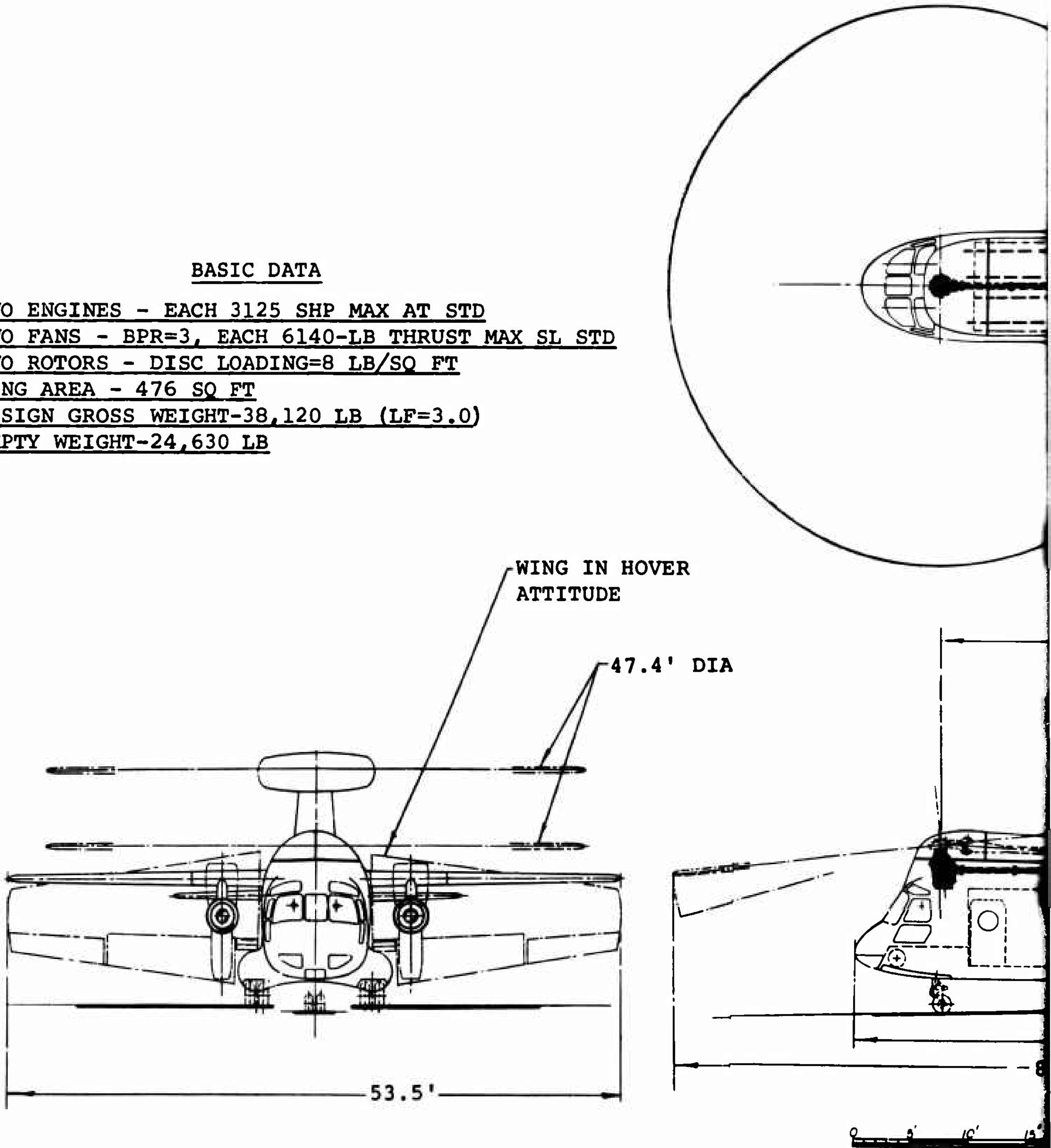
BASIC DATA

1. TWO ENGINES - EACH 3125 SHP MAX AT STD
2. TWO FANS - BPR=3, EACH 6140-LB THRUST MAX SL STD
3. TWO ROTORS - DISC LOADING=8 LB/SQ FT
4. WING AREA - 476 SQ FT
5. DESIGN GROSS WEIGHT-38,120 LB (LF=3.0)
6. EMPTY WEIGHT-24,630 LB



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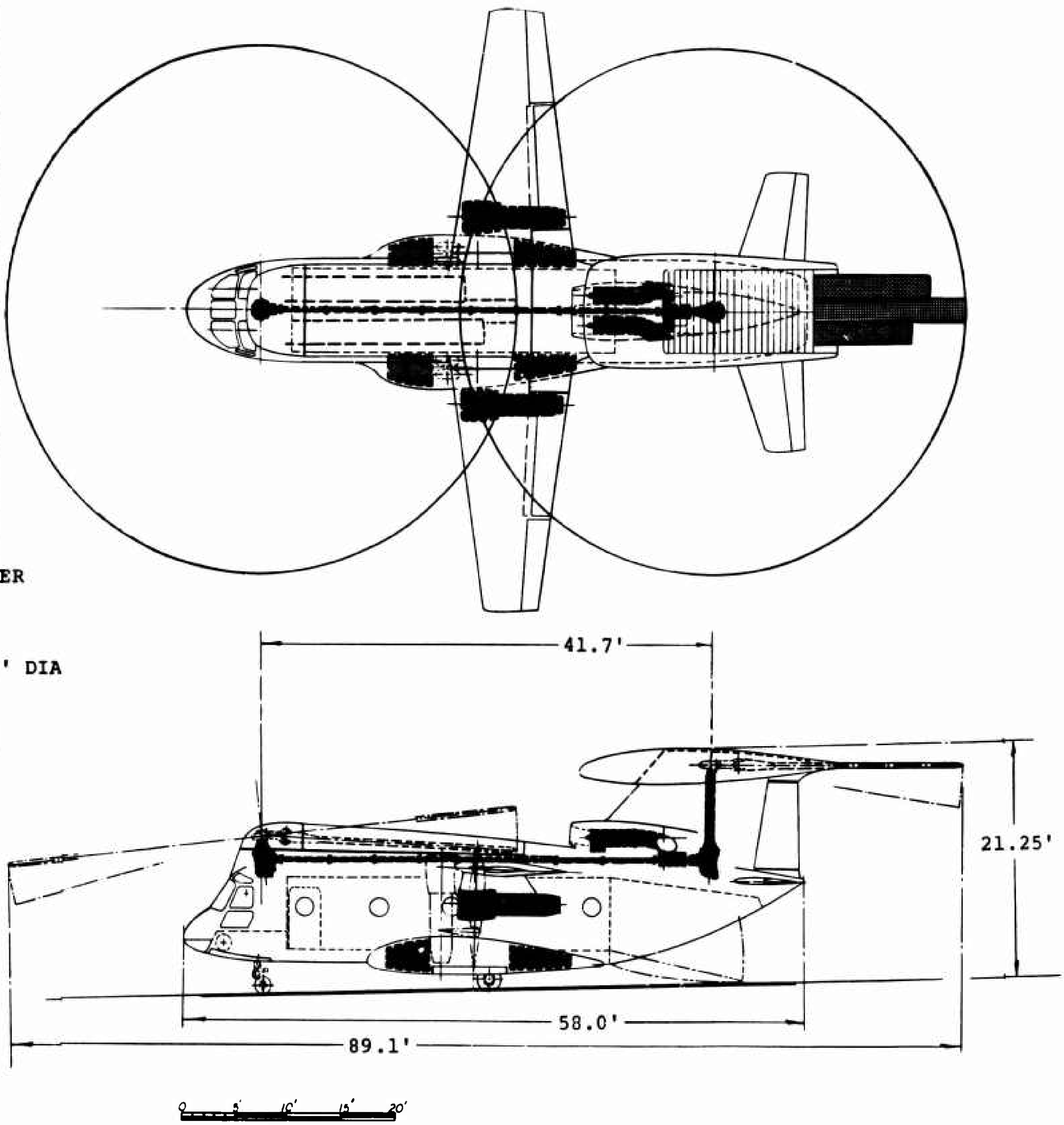

Figure 178. Tandem Rotor Composite Aircraft Propulsion System 3, Tilting Fans

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13. ABSTRACT
This report examines the feasibility and relative merit of five gas- and shaft-coupled cruise fan propulsion systems in various compound and composite aircraft configurations. A 1970 time period is assumed. Propulsion and airframe parameters are defined, and the results of an optimization process for maximum aircraft relative productivity in a fixed short-range transport mission are shown. Ranges of vehicle disc loading of from 5 to 11 pounds per square foot and of fan bypass ratio of from 3 to 12 are covered. The problems of rotor-fan power transfer systems are reviewed and possible solutions evaluated. Detailed weight data is given for optimum aircraft and propulsion system combinations. Convertible shaft-driven cruise fans in an integrated propulsion system are found to be particularly attractive; however, considerable work is required in development of power management systems.

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